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The Journal of the American Rocket Society is devoted to the advancement of the field of jet propulsion through the publication of original papers disclosing new knowledge and new developments. The term "jet propulsion" as used herein is understood to embrace all engines that develop thrust by rearward discharge of a jet through a nozzle or duct, and thus includes systems utilizing atmospheric air and underwater systems, as well as rocket engines. The Journal is open to contributions, either fundamental or applied, dealing with specialized aspects of jet and rocket propulsion, such as fuels and propellants, combustion, heat transfer, high temperature materials, mechanical design analyses, flight mechanics of jet-propelled vehicles, astronautics, and so forth. The Journal endeavors, also, to keep its subscribers informed of the affairs of the Society and of outstanding events in the rocket and jet propulsion field.

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JOURNAL OF THE AMERICAN ROCKET SOCIETY

Editor-in-Chief

MARTIN SUMMERFIELD

The 1951 ARS Annual Convention: A Technical Summary

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General Impressions

THE presentation of sixteen technical papers of widely varying scope highlighted the 1951 Annual Convention of the American Rocket Society held at Chalfonte-Haddon Hall in Atlantic City, N. J., over the three-day period from November 28 to November 30. A record attendance that represented a nation-wide interest assured the most effective exchange of technical information both through formal and informal discussions. The broadening interests of the Society were indicated by papers concerned with such subjects as the experimental determination of aerodynamic characteristics of bodies in free flight, problems of stabilization of supersonic vehicles, and a discussion of methods of producing ultra-high exhaust velocities in rocket motors by use of schemes outside the areas of normal chemical reactions. Most papers, however, were primarily of a descriptive nature, and although these presentations on the whole were both interesting and entertaining, it is somewhat unfortunate that a deeper treatment from a technical standpoint of some of the multitudinous problems facing the rocket and jet propulsion engineer was not possible. The presentation of even very rough and preliminary analytical approaches to a problem, with resultant discussion and interchange of ideas in a group of such unique yet broad character, can frequently lead to most fruitful and productive results. It is to be hoped that such presentations may be encouraged in future meetings together with an emphasis on open discussions. Extension of available time for the latter at the expense of a verbatim offering from a prepared manuscript would generally prove more effective.

First Session

Five sessions comprised the 1951 Annual Meeting. The first session on the afternoon of Wednesday, November 28, was under the capable chairmanship of C. C. Ross of the Aerojet Engineering Corporation, assisted by Powel Brown of the M. W. Kellogg Company, as vice-chairman. The initial paper of the program was a joint contribution by C. W. Besserer and A.J. Bell of The Johns Hopkins University Applied Physics Laboratory. A survey of the very important problem of directional attitude stabilization of supersonic vehicles was presented. An objective discussion of intelligence and force-controlling systems included examples of two typical jet vane systems using a gyroscope reference system for intelligence. In one system a mechanical servo was used, while in the other a conventional electro-hydraulic servo was employed. Engineering problems in the design of stabilization systems were pointed out. The supersonic vehicle represents a system having little inherent damping, and a system for stabilization demands a fast-acting, well-matched servo. The design of a satisfactory servo loop requires the adjustment of the basically contradictory requirements, speed of response, and dynamic stability. The paper limited itself to the directional attitude stabilization case. Roll attitude stabilization was not discussed.

The second paper of the afternoon dealt with experimental dynamic launching techniques for testing aircraft rockets. The facilities at the Naval Ordnance Test Station at Inyokern, Calif., discussed by A. W. Nelson, permit the carrying-out of terminal ballistic tests and firing of rockets under conditions simulating operations at supersonic speeds. Rocket-accelerated launcher carriages and test vehicles traveling on long fixed tracks allow studies under the simulated conditions of practical interest. The vehicles are mounted on aluminum or magnesium skids that slide on rails. Five railed tracks are presently available for such studies at Inyokern. They range from a horizontal 2-mile railroad type track to a 550-ft track inclined at 6 degrees. Such test techniques provide controlled conditions so that cameras and other instruments can be synchronized with a predetermined position, velocity, or acceleration.

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T. F. Reinhardt of the U. S. Naval Rocket Test Station at Lake Denmark, N. J., presented the final paper of the afternoon entitled, "Unusual Applications of the Momentum Principle." The study dealt with schemes for achieving extremely high values of specific impulse in rocket motors, above those possible from ordinary chemical reactions. The point was made that high specific impulse and high efficiency are not synonymous since the power required per unit of thrust, defined as the heat content of the working gas divided by the specific impulse, varies proportionally to the specific impulse. Schemes discussed included the use of a light beam, which, however, would require 1,330,000 kilowatts to produce 1 lb of thrust; the use of an electron beam affords another means. We find, in this case, that high specific impulses demand an astronomical power consumption. If we attempt to get the same thrust at lower impulse values, the current requirements mount astronomically. The use of a condenser arrangement which could be charged over a long period was also discussed. Finally the practical possibilities of using a working fluid receiving its energy from another power source such as a nuclear reactor was reviewed. Various working fluids were compared; temperature limitations and other related engineering problems were mentioned. Open discussion appeared to confirm that conventional chemical fuel rockets are not due to be outmoded in the immediate future.

Second Session

The Thursday morning meeting found J. H. Sheets of the Curtiss-Wright Corporation as presiding officer, with M. Meyer of the same firm as vice-chairman. The introductory paper was a presentation by Frank A. Coss of Reaction Motors, Inc., on problems of installation of rocket engines in airplanes. Methods of mounting, engine-to-airframe sealing, separation of power plant and airframe components, ambient temperature and pressure considerations, design of propellant lines and feed systems were included. Installations have been made to date only on research airplanes. Simplified control systems, probably even a simple, single throttle can be evolved for operational airplanes.

The second paper of the morning was a description of the Naval Ordnance Test Station Aeroballistics Laboratory, by I. E. Highberg of the NOTS staff. The facility is designed for the experimental determination of the aerodynamic and ballistic characteristics of rocket models. Free-flight testing affords a simpler and less expensive means than the wind tunnel for getting special information, such as determination of the dynamic parameters. The wind tunnel techniques to obtain such information are extremely difficult, although the tunnel constitutes the usual approach to the measurement of the so-called static parameters or coefficients. Aeroballistic ranges normally employ shadowgraph methods and associated spark photography to obtain the pictorial information. Frequently the shadow is thrown onto a beaded screen, which is photographed. A new technique for obtaining photographs was developed at NOTS. Investigation proved the feasibility of obtaining several silhouette images on each plate. "Scotchlite" reflex reflective sheeting is used for the silhouette background. The technique has been developed so that six image photographs can be assessed with a mean deviation for the comparator readings of only a few microns. Special timing currents have been developed to produce an accurately timed, microsecond duration, flash illumination synchronized with the passage of the rocket model. A query as to the feasibility of using live rockets for propulsion in the range was answered by the statement that, at present, too great inaccuracies exist in the trajectory course, and also that the photographic technique would have to be revised in view of the illumination that would be produced by the rocket exhaust.

The final paper of the second session was a description of the United States Air Force Experimental Rocket Engine Test Station at Edwards Air Force Base, California, by R. A. Schmidt and Donald L. Dynes. The role of the station is to follow up the power plant development at the contractor's laboratory with complete tests of the motor and its components; to provide for further development at this large, well-instrumented test station; to determine endurance, suitability, and compliance with military specifications; and to perform final tests on a missile mock-up. The station is thus to serve as a central, common facility for private contractors. The laboratory is designed to provide great versatility in propellant utilization, thrust ratings, engine mountings, and instrumentation. The engines to be tested and developed will include rocket engines for missiles, aircraft, and assisted take-off units.

Third Session

The presiding officers at the third session of the Convention were N. C. Appold and E. N. Hall of the Power Plant Laboratory, Wright Air Development Center, Dayton, Ohio.

The opening paper by J. A. Bierlein of the Air Research Command, U.S.A.F., and Karl Scheller of Wright-Patterson Air Force Base, was a study of the effect on performance of jet engines by the use of an isothermal expansion process rather than an adiabatic process. The detailed flow characteristics using the equations of Hawthorne and Shapiro were calculated for two cases: One where all the propellants are introduced in the chamber and combustion proceeds throughout the nozzle in the prescribed fashion; the other where the propellants are introduced continuously in such a way as to produce the isothermal flow conditions. The obvious thermodynamic result of an inferior efficiency as compared to the adiabatic case was established. It was pointed out that the isothermal process might lead to improved performance, however, in the event that thermal limitations of structural materials preclude the

adiabatic utilization of the total energy available to the working fluid. In the discussion, L. Crocco pointed out that even in the case of an effectively unlimited energy supply, the process of transferring the required energy to the working fluid under the high-temperature, high-velocity condition implied would be highly inefficient.

An approximate theory of porous, sweat, or film cooling with reactable fluids was presented by L. Crocco, of Princeton University, in the second paper of the afternoon. The theoretical treatment of the porous cooling problem by W. D. Rannie, and its application to sweat or film cooling as pointed out by J. Sloop, were modified by Crocco to include the very important extension to the case of a reactive coolant. Previous studies had been confined to the case of an inert coolant. In addition, Rannie's restrictive assumption of the independency of the properties of the coolant and the combustion gases on temperature is shown to be unessential. Dropping of this restriction adds no complication to the analysis. Among the important additional assumptions made to treat the case of the reactive coolant are those of diffusion of the combustion gases as a whole, the extension of the Reynolds analogy to mass transport and hence turbulent diffusion of chemical species, and the hypothesis that the reaction times of the mixtures are short with respect to the other times involved so that at every layer the mixture can be considered to immediately reach the final condition. In the laminar sublayer, where the oxygen concentration will be extremely low, ordinary thermodynamic calculations for determination of the products of combustion are impotent and assumptions must be made. Although the analysis cannot be expected to yield highly accurate required coolant rates, the paper is an important contribution to an understanding of the phenomena involved and will predict trends and orders of magnitude.

J. B. Hatcher and D. R. Bartz of the Jet Propulsion Laboratory, California Institute of Technology, offered the next paper of the afternoon dealing with high-flux heat transfer to JP-3 and RFNA and coke deposition of JP-3. Data were presented for heat transfer in the forced convection and the nucleate-boiling regions up to the maximum flux limits of 7 to 10 BTU/in.^{2}/sec. The pressure range from 50 to 500 psia was covered. Simple expressions for coke deposition with JP-3 were found and utilized to give correction to clean tube values.

The final paper of the session was a presentation by W. F. Kaufmann and B. N. Abramson of the U. S. Naval Air Rocket Test Station, dealing with an idea to use various forms of concentric nozzles to achieve a shortening of a rocket exhaust nozzle keeping the angle of divergence of each portion of the nozzle at the single nozzle value, normally a half angle of 15°. Some confusion appeared to exist as to the reason for maintaining this value for the divergence angle. This may possibly be due to erroneous ideas on flow separation in nozzles.

Fourth Session

The papers presented on the last day of the Convention were, with one exception, concerned at least in some measure with the problem of combustion instability in rocket motors. The morning session was devoted to photographic techniques of internal processes within the combustion chamber, while the afternoon session attempted to shed more light on the role of the injector head itself as a cause of combustion instability.

The morning session was opened by C. M. Hudson of the Office of Chief of Ordnance, Washington, D. C. He was assisted by H. F. Calcote of Experiment, Incorporated, Richmond, Va.

The first paper, one of the most interesting of the meeting, was "Combustion Studies with a Rocket Motor Having a Full Length Observation Window." It was prepared by Kurt Berman and Stanley Logan, of the General Electric Malta Test Station. The authors described the use of a rocket motor with an observation slit in the side of the combustion chamber as a means for recording on film the path of radiating particles in the combustion chamber. Pictures were shown representing the ignition, preliminary, and full-stage combustion phases during both stable and unstable operation.

The test motor was a 1200-lb thrust rocket using ethyl alcohol and liquid oxygen as propellants. The observation window was a $^1/_4$-in. slit made of two quartz plates $^1/_2$ in. thick. Nitrogen gas was admitted between the quartz plates and also on the inner side of the plates for cooling purposes. Three different types of injector heads were investigated in stable and unstable operation. A continuous strip camera was used to record the motor radiation.

The results of these tests were recorded on ordinary and colored film. Radiation streaks on the film were used to measure gas velocity which, in turn, roughly indicated the reaction zones. Streak spacing, for both stable and unstable operation, did not exhibit any regular frequency or repetitive pattern.

During unstable operation, high-frequency oscillations were observed and were in good agreement with calculated "organ pipe" frequencies. Perhaps the most unusual result of these tests was the observation of intermediate frequency oscillations on the order of 200–300 cps. The authors attributed this phenomenon to liquid and injector system characteristics. A theory for high-frequency instability was also suggested.

In the discussion period which followed, prepared comments were presented by L. Crocco of Princeton University. He pointed out that the assumption that the film indicated two-dimensional effects was optimistic, particularly during the high-frequency unstable operation. Dr. Crocco also offered another explanation for high-frequency oscillations, based upon his recent

work (see pages 7–16). Finally he suggested that the intermediate oscillation frequencies observed by the authors appeared too high to be classed with the usual "chugging" phenomena, and that certain measured gas velocities appeared rather low and should be checked with specific impulse calculations.

The next paper also dealt with photographic combustion studies. The title was "Photographic Techniques Applied to Combustion Studies—Two-Dimensional Transparent Thrust Chamber," by John H. Altseimer of Aerojet Engineering Corporation. During the presentation the audience was treated to some very striking motion pictures, mostly in color, of a transparent rocket motor in operation.

The motor was constructed of Lucite plates and represented a 0.47-in. axial slice of a 1000-lb rocket. The operating chamber pressure was 300 psia. Five different injector heads were investigated.

Very definite combustion patterns were established by each injector head type. Perhaps the most interesting visual phenomenon was the appearance of cylindrical flame fronts, parallel to the axis of the combustion chamber. Particularly noticeable for repetitive pattern injectors, these striations could be traced the length of the combustion chamber and into the exhaust nozzle. Low-frequency unstable operation was also recorded, and a frame-by-frame analysis of the radiation fluctuations was compared with pressure fluctuations. The author made several recommendations for a future investigation using this technique which he hoped would lead to even more definite conclusions.

The last paper of the morning session was a general discussion of "Experimental Problems in High Pressure Combustion" by R. L. Wehrli of Reaction Motors, Inc. Starting with a brief mention of the advantages of high-pressure combustion systems, the author outlined three solutions to the problem of cooling the gas generator in high-temperature reactions. Following a few brief remarks on high pressure seals, Wehrli discussed the important problems of the instrumentation of high temperature and high pressure reactions. In particular, he pointed out the usefulness of the common strain gage in measuring pressures. Concerning the difficult problem of flow measurements at high pressures, the rotary vane fluid-velocity meter was mentioned as a possible solution for certain applications.

In the short open period at the end of the paper, various instruments were proposed as being applicable to such a problem, the most notable among them being the electromagnetic flowmeter and the cavitating Venturi for flow measurement at high pressures.

Fifth Session

The chairman of the afternoon session was George P. Sutton from North American Aviation, Inc. He was assisted by a colleague, O. K. Doyle, from the same company.

As a contribution to the general problem of rocket combustion stability, R. P. Northup of the General Electric Malta Test Station, presented a paper called "Flow Stability in Small Orifices." He described a series of experiments attempting to determine the effect of the diameter, length, and shape of the orifice; the effect of the composition and cross velocity of the liquid; and the effect of the pressure and density of the gas into which the liquid discharges.

From the results of these tests it is seen that for sharp-edged orifices, certain pressure ranges should be avoided. In general, higher pressure drops tend to provide more stable flow. The exact pressure ranges for desirable flow can be predicted only when the density of the receiving atmosphere is known. The author also stated that flow from orifices with wetted walls is inherently unstable. He concluded that for best results, cross velocity and turbulence behind the orifice should be minimized, and that if low pressure drops are necessary, some orifice other than a sharp-edged orifice should be used.

The second paper of the afternoon was prepared by Kurt R. Stehling of the Bell Aircraft Corporation, and dealt with the photographic analysis of injector sprays. Mr. Stehling described the major types of rocket injector heads and their variations, and discussed briefly the effect of the injector and its design on rocket performance. The author considered the problem of spray analysis and the various factors affecting the spray shape. Some remarkable colored photographs showed the hydraulic testing of various types of injector heads. A blue fluid representing the oxidizer and a yellow fluid representing the fuel were used, and high-speed stroboscopic photographs were taken of the injection process. The general types of injectors considered were: (1) random showerhead, (2) splash, (3) impinging, (4) hypoid, (5) concentric ring, and (6) commercial spray nozzle. Mr. Stehling also mentioned the possibility that gas pockets in the injector head manifold behind the orifices could contribute to unstable combustion characteristics, and explained his point with colored photographs of an injector with a transparent back plate. A few remarks about the phenomenon of "hydraulic flip," illustrated by photographs, closed the paper.

The final paper of the 1951 Convention, entitled "Fluctuations in a Spray Formed by Two Impinging Jets," was prepared by Marcus F. Heidmann and Jack C. Humphrey of the NACA Lewis Flight Propulsion Laboratory.

In this paper, the feedback loop analogy was used to describe low-frequency combustion instability, and the requirements for the extension of the analogy to high-frequency "screaming" instability were summarized. The authors advanced the theory that perhaps injector spray fluctuations could provide the additional gain factor necessary to explain high-frequency oscillations in terms of the feedback loop.

(Continued on page 27)

Aspects of Combustion Stability in Liquid Propellant Rocket Motors[1]

Part II: Low Frequency Instability with Bipropellants. High Frequency Instability

By L. CROCCO[2]

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In Part I the problem of low-frequency combustion instability in monopropellant rocket motors has been analyzed under the basic assumption that the time lag is affected by pressure variations. The same idea is applied now to the bipropellant case. It is shown that for certain relations between the parameters of the two feeding systems, the results of Part I on monopropellant systems can be rigorously applied. The effect of varying the parameters from these values is also examined. A brief discussion is given of a secondary effect neglected in Part I, applicable to both monopropellant and bipropellant cases. Finally, using a simplified model for the spatial distribution of the combustion in the chamber, the case of high-frequency instability is analyzed. The correlation between unstable and natural frequencies is shown, and the ranges of the time lag in which high-frequency modes become unstable are found. The generalization of the results to real models is discussed.

Nomenclature

Combustion Chamber

t = time
τ = instantaneous value of the time lag
$\bar{\tau}$ = value of the time lag in steady operation
n, m = pressure exponents of pressure dependence of the processes taking place during the time lag
p = instantaneous pressure in the combustion chamber
$\bar{p}$ = pressure in the combustion chamber in steady operation
φ = $(p - \bar{p})/\bar{p}$ = fractional variation of pressure in the combustion chamber
$\dot{m}_i$, $\dot{m}_b$, $\dot{m}_e$ = instantaneous rate of injection, burning and ejection of propellants
$\bar{\dot{m}}$ = common value of these in steady operation
μ = $(\dot{m}_i - \bar{\dot{m}})/\bar{\dot{m}}$; $\mu_b = (\dot{m}_b - \bar{\dot{m}})/\bar{\dot{m}}$ = fractional variation of injection and burning rate
M_g = instantaneous mass of gases in the combustion chamber
$\bar{M}_g$ = same in steady operation
θ_g = $M_g/\bar{\dot{m}}$ = gas residence time in steady operation
θ_t = $\theta_g + \bar{\tau}$ = total residence time of propellants in steady operation
c^*, L^* = characteristic velocity and length of the rocket
R, γ = gas constant and adiabatic index of combustion gases
T_g, ρ_g = absolute temperature and density of combustion gases
$z = t/\theta_g$; δ = $\bar{\tau}/\theta_g$ = reduced time and reduced time lag
α = $\lambda + i\omega$ = root of the characteristic equation with the reduced time as the independent variable
λ = reduced amplification coefficient
ω = reduced angular frequency
Ω = angular frequency
T = period of oscillations
Subscript $_*$ = critical conditions of incipient instability

Monopropellant Feeding System

p_1 = instantaneous pressure at that place in the feeding line where the capacitance representing the elasticity is located.
$\bar{p}_1$ = same in steady operation
Δp = $\bar{p}_1 - \bar{p}$ = injector pressure drop in steady operation
ψ = $(p_1 - \bar{p}_1)/2\Delta p$ = relative variation of p_1
p_0 = regulated gas pressure for constant pressure supply
A_i, A_l = area of the injector ports and of the cross section of the feeding line
v_i, v_l = velocity of the fluid in A_i and A_l
ρ = density of the propellant
C = variable volume of the capacitance introduced in the feeding system to represent its elasticity
χ = dC/dp = compressibility coefficient
M_l = mass of propellant in the line
l = length of the line
ξ = fractional length for the constant pressure supply
E = $2\rho\chi\Delta p/\bar{\dot{m}}\theta_g$ = elasticity parameter of the line
J = $l\bar{\dot{m}}/2\Delta p A_l\theta_g$ = inertia parameter of the line
P = $\bar{p}/2\Delta p$ = pressure drop parameter
$\dot{m}_1$, μ_1 = instantaneous mass flow and its fractional variation upstream of the capacitance

Nonuniform Time Lag

$f (0 < f < 1)$ = location of a given fraction of propellant in the jet from the injector
$\tau(f)$ = time lag corresponding to the fraction considered
$\delta(f)$ = corresponding reduced time lag
δ_m = average reduced time lag
$\epsilon(f)$ = $\delta(f) - \delta_m$ = variation from the average
ϵ_0, ϵ_1 = extreme values of $\epsilon(f)$

Received July 3, 1951.

[1] Part I of this paper appeared in the preceding issue of the JOURNAL, November 1951.

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CORRECTION: In Part I, page 163, November 1951, the latter part of Equation [3.2] should read:

$$\int_{t-\tau_1}^{t} p^m dt' = \bar{\tau}_1 \bar{p}^m = \text{const.} \qquad [3.2]$$

Additional Nomenclature for Part II

Bipropellant System

r	= instantaneous mixture ratio
$\bar{r}$	= mixture ratio in steady operation
K	= coefficient related to the variation of temperature with mixture ratio
V	= total volume of the combustion chamber
dV	= volume element of the combustion chamber
H	= $(\bar{r} - 1)/2(\bar{r} + 1)$ = coefficient representing the deviation from unity of the mixture ratio
q, j	= fractional variation of P, J of the two feeding systems with respect to the average value

Subscripts

i	= injector end
e	= exhaust end
o	= oxidizer
f	= fuel

High-Frequency Oscillations

$u(x, t)$, $\bar{u}$	= local instantaneous and steady operation velocity of the gases
$\rho_g(x, t)$, $\bar{\rho}_g$	= same for density
$c(x, t)$, $\bar{c}$	= same for sound velocity
$p(x, t)$, $\bar{p}$	= same for pressure
$M(x, t)$, $\bar{M}$	= same for Mach number
$\nu(x, t)$, $\sigma(x, t)$, $\varphi(x, t)$	= fractional variation of velocity, density, pressure with respect to steady operation values
A_c	= area of the cross section of the combustion chamber
L	= length of the combustion chamber
$B(\bar{M})$	= $\left(1 + \frac{\gamma - 1}{2}\bar{M}\right)\Big/\left(1 - \frac{\gamma - 1}{2}\bar{M}\right)$
Θ	= characteristic time of the combustion chamber
α	= $\Lambda + i\Omega$ = complex root of the characteristic equation
Λ	= amplification coefficient
Ω	= angular frequency
h, k	= integers characterizing the oscillation modes

10 Bipropellant Systems

The presence of two propellants, and of two feeding systems, introduces two complications in the problem. First, the additional degrees of freedom of the second feeding system add a certain number of supplementary unknowns and equations; second, due to the eventuality of different fractional variations of the two rates of injection, the temperature of the gases can vary with time and space. For a given set of propellants, the temperature of the combustion gases is a function of the local mixture ratio, $r = O/F$ between the oxidizer and the fuel. If there is a small variation of the mixture ratio with respect to the one of the steady case, $\bar{r}$, the fractional variation of temperature is also small and can be expressed as

$$\frac{T_g - \bar{T}_g}{\bar{T}_g} = 2K\frac{r - \bar{r}}{\bar{r}}, \text{ with } 2K = \frac{\bar{r}}{\bar{T}_g}\frac{dT_g}{dr} \quad \text{[10.1]}$$

the derivative being computed at the operating point. K is therefore zero only if the rocket operates at the conditions of maximum temperature, which is not the case in general. In deducing the equation of the combustion chamber we have to take into account this temperature variation, so that instead of Equations [4.1] we write:

$$\frac{\dot{m}_e}{\bar{\dot{m}}} = \frac{p}{\bar{p}}\left(\frac{\bar{T}_g}{T_{ge}}\right)^{1/2}; \quad \frac{M_g}{\bar{M}_g} = \frac{1}{\bar{\rho}_g V}\int^e \rho_g\, dV = \frac{p}{\bar{p}}\frac{\bar{T}_g}{V}\int^e \frac{dV}{T_g} \quad \text{[10.2]}$$

the exhaust rate being inversely proportional to the square root of the instantaneous exhaust temperature T_{ge}, and the mass contained in the combustion chamber being obtained by a volume integration of the local, instantaneous density of the gases ρ_g from the injection to the exhaust section.

In order to evaluate the quantities of Equation [10.2], some assumptions are necessary on the way the combustion takes place. So far, the gas residence time was introduced only as the ratio $\bar{M}_g/\bar{\dot{m}}$, without attributing to it any physical meaning. It is doubtful if, in fact, θ_g can really represent the time of residence of all the particles in the combustion chamber, since probably this quantity is different for different portions (due to recirculation effects) and θ_g represents only an average value. Nevertheless, let us make the rough but simple assumption that θ_g is the residence time of all the particles and that during this time every particle travels from the injector end, where burning takes place, to the exhaust end, carrying with it the temperature developed at the combustion instant; that is, the temperature corresponding to the mixture ratio τ seconds earlier. Then T_{ge} at the time t will correspond to the mixture ratio of the propellants injected at the time $t - \theta_t = t - \bar{\tau} - \theta_g$ (Equation [4.6]), or at the reduced time $z - \delta - 1$. If we introduce the oxidizer and the fuel injection rates and their fractional variations,

$$\dot{m}_o = \bar{\dot{m}}_o(1 + \mu_o); \quad \dot{m}_f = \bar{\dot{m}}_f(1 + \mu_f)$$

the mixture ratio is given by

$$r = \frac{\dot{m}_o}{\dot{m}_f} = \frac{\bar{\dot{m}}_o}{\bar{\dot{m}}_f}(1 + \mu_o - \mu_f) = \bar{r}(1 + \mu_o - \mu_f) \quad \text{[10.3]}$$

so that, using Equation [10.1] and the superscript convention of Equation [4.2]:

$$\frac{T_{ge}}{\bar{T}_g} = 1 + 2K(\mu_o - \mu_f)^{(\delta + 1)} \quad \text{[10.4]}$$

Hence, from Equation [10.2] the fractional variation of the exhaust rate is given, for small variations, by

$$\mu_e = \frac{\dot{m}_e - \bar{\dot{m}}}{\bar{\dot{m}}} = \varphi - K(\mu_o - \mu_f)^{(\delta + 1)} \quad \text{[10.5]}$$

The Equation [4.3] of the mass balance can be rewritten in the form

$$\frac{d}{dz}\left(\frac{M_g}{\bar{M}_g}\right) + \mu_e = \mu_b \quad \text{[10.6]}$$

The first term can be obtained as follows, with two different assumptions on the temperature distribution through the combustion chamber.

Assumption (*a*): If the average temperature of the gases in the chamber is assumed to be independent of the time, then

$$\frac{M_g}{\bar{M}_g} = \frac{p}{\bar{p}} = 1 + \varphi \ldots\ldots\ldots\ldots [10.7]$$

which is the same as Equation [4.1].

Assumption (b): If the temperature variations are not neglected, we can write in the last Equation [10.2]

$$\frac{\bar{T}_g}{V}\int_i^e \frac{dV}{T_g} = 1 + y \ldots\ldots\ldots\ldots [10.8]$$

y being a small quantity, so that

$$\frac{M_g}{\bar{M}_g} = 1 + \varphi + y, \text{ and: } \frac{d}{dz}\left(\frac{M_g}{\bar{M}_g}\right) = \frac{d\varphi}{dz} + \frac{dy}{dz} \ldots [10.9]$$

We have then to compute the time derivative of y. In the aforesaid assumption that all of the combustion takes place practically at the injector end, and that each slice of gas preserves its temperature (independently of the pressure oscillations[3]) up to the exhaust end, we see that the only variation of the integral of Equation [10.8] is due to the rate of variation of the integrand at the two ends. Moreover, in the aforesaid assumptions, the volume displacement at every point is proportional to the time, so that:

$$\frac{dV}{dt} = \left(\frac{dV}{dt}\right)_i = \left(\frac{dV}{dt}\right)_e = \frac{V}{\theta_g} \ldots\ldots\ldots [10.10]$$

and

$$\frac{d}{dt}\left(\frac{\bar{T}_g}{V}\int_i^e \frac{dV}{T_g}\right) = \frac{\bar{T}_g}{V}\left[\frac{1}{T_{ge}}\left(\frac{dV}{dt}\right)_e - \frac{1}{T_{gi}}\left(\frac{dV}{dt}\right)_i\right] = \frac{1}{\theta_g}\left(\frac{\bar{T}_g}{T_{ge}} - \frac{\bar{T}_g}{T_{gi}}\right)$$

and therefore

$$\frac{dy}{dz} = 2K\left[(\mu_o - \mu_f)^{(\delta+1)} - (\mu_o - \mu_f)^{(\delta)}\right] \ldots [10.11]$$

$T_{ge}/\bar{T}_g$ being given by Equation [10.4] and $T_{gi}/\bar{T}_g$ by Equations [10.1] and [10.3] computed at the end of the time lag.

Finally, the fractional variation of the burning rate is given by Equation [4.9]:

$$\mu_b = \mu^{(\delta)} + n\left(\varphi - \varphi^{(\delta)}\right) \ldots\ldots\ldots\ldots [10.12]$$

where the fractional variation of the total injection rate, μ, is determined in the following way. From the equation:

$$\dot{m} = \bar{\dot{m}}(1 + \mu) = \dot{m}_o + \dot{m}_f = \bar{\dot{m}}_o + \bar{\dot{m}}_f + \bar{\dot{m}}_o\mu_o + \bar{\dot{m}}_f\mu_f$$

as

$$\bar{\dot{m}} = \bar{\dot{m}}_o + \bar{\dot{m}}_f = \bar{\dot{m}}_f(1 + \bar{r}) = \bar{\dot{m}}_o(1 + \bar{r})/\bar{r}$$

we deduce

$$\mu = \frac{\bar{r}\mu_o + \mu_f}{\bar{r} + 1} = (1/2 + H)\mu_o + (1/2 - H)\mu_f, \ldots [10.13]$$

with

$$H = \frac{\bar{r} - 1}{2(\bar{r} + 1)} \ldots\ldots\ldots\ldots [10.14]$$

representing the deviation from unity of the mixture ratio.

[3] See Section 11 on page 11.

Substituting now in Equation [10.6] from Equations [10.9] and [10.11], in the first term, Equation [10.5] in the second term, and Equations [10.12) and [10.13] in the right-hand side, we obtain the equation of the combustion chamber, with the conditions of assumption (b):

$$\frac{d\varphi}{dz} + 2K\left[(\mu_o - \mu_f)^{(\delta+1)} - (\mu_o - \mu_f)^{(\delta)}\right] + \varphi - K(\mu_o - \mu_f)^{(\delta+1)} = (1/2 + H)\mu_o^{(\delta)} + (1/2 - H)\mu_f^{(\delta)} + n\left(\varphi - \varphi^{(\delta)}\right) \ldots [10.15]$$

If we choose the assumption (a), the equation is the same, except that the term with coefficient $2K$ disappears. In both cases, Equation [10.15] contains now three unknown variables φ, μ_o, μ_f; the additional equations are provided by the feeding systems. If we assume for simplicity a constant pressure supply, without elasticity, for both propellants, the corresponding equations are the same as Equation [6.8] with $\psi = 0$:

$$\left.\begin{aligned} P_o\varphi + J_o\frac{d\mu_o}{dz} + \mu_o &= 0 \\ P_f\varphi + J_f\frac{d\mu_f}{dz} + \mu_f &= 0 \end{aligned}\right\} \ldots\ldots\ldots [10.16]$$

the coefficients being defined for the two lines by Equation [6.7]. P_o and P_f are different if the two feeding systems have different pressure drops; also J_0 and J_f can, in general, be different; and we can define mean values P, J and fractional deviations, q, j, from the mean values in the following way

$$P_o = (1 - q)P;\; P_f = (1 + q)P;\; J_o = (1 - j)J;\; J_f = (1 + j)J \ldots [10.17]$$

As in previous sections, let us try a solution where φ, μ_o, μ_f are proportional to $\exp(\alpha z)$; replacing in Equations [10.15], [10.16] we see that the solution is possible only if

$$\begin{vmatrix} (1 - q)P & 1 + (1 - j)J\alpha & 0 \\ (1 + q)P & 0 & 1 + (1 + j)J\alpha \\ 1 - n + \alpha + ne^{-\alpha\delta} & -(g + 1/2)e^{-\alpha\delta} & (g - 1/2)e^{-\alpha\delta} \end{vmatrix} = 0$$

where

$$g(\alpha) = H + Ke^{-\alpha}$$

in the assumption (a) (Equation [10.7]), and

$$g(\alpha) = H + K(2 - e^{-\alpha})$$

in the assumption (b) (Equations [10.8], [10.9]).

Restricting our analysis to the research of critical conditions we take $\alpha = i\omega$, and developing the determinant we have the complex equation

$$\left[(1 - n + i\omega_*)e^{i\omega_*\delta_*} + n\right]\left[1 - (1 - j^2)J^2\omega_*^2 + i2J\omega_*\right] = -P\{1 - 2qg(i\omega_*) + iJ\omega_*\left[1 + 2(j - q)g(i\omega_*) - jq\right]\} \ldots [10.18]$$

which corresponds to two real equations, sufficient for the determination of ω_* and δ_*. The function $g(i\omega_*)$ can be written as

$$g(i\omega_*) = H + K \mp K(1 - \cos\omega_*) \mp iK\sin\omega_* \ldots [10.19]$$

with the upper sign corresponding to assumption (a) and the lower to assumption (b).

A general discussion of the solutions of Equation

[10.18] seems to be too complicated due to the number of parameters involved (n, P, J, q, j, H, K). Let us make only a few remarks sufficient to show some of its peculiarities. First, suppose $P_o = P_f$, and $J_o = J_f$, so that $q = j = 0$. In this case Equation [10.18], divided by the nonzero factor $1 + iJ\omega_*$, is reduced to what becomes Equation [8.1] with $E = 0$. We can therefore conclude that the bipropellant system is entirely equivalent to a monopropellant system when the two feeding systems have the same Δp and the same J. Equation [6.7] shows that the last condition is verified, provided the first is too, if the products of the lengths of the lines and the corresponding mass velocities are identical. The equivalence to the monopropellant system does not imply that the stability conditions are the most favorable. Suppose, in fact, we start from $q = j = 0$; let us examine the effect of small variations of these quantities on δ_*. For this purpose it is sufficient to evaluate the derivatives of δ_* with respect to q and j at $q = j = 0$. We will consider only the two extreme cases of very short lines and very long lines. In the first case, $J \cong 0$ and Equation [10.18] is simplified to

$$(1 - n + i\omega_*)e^{i\omega_*\delta_*} = -\{n + P - 2q\,[H + K \mp K(1 - \cos\omega_*)] \pm i2qK\sin\omega_*\} \quad [10.20]$$

When $q = 0$ this is reduced to

$$(1 - n + i\omega_*)\,e^{i\omega_*\delta_*} = -(n + P) \quad [10.21]$$

whose solution is

$$\omega_* = \sqrt{(n + P)^2 - (1 - n)^2};$$

$$\omega_*\delta_* = \pi - \tan^{-1}\frac{\omega_*}{1 - n} \quad [10.22]$$

As j has no influence on Equation [10.20], we differentiate only with respect to q. Performing the differentiation at $q = 0$, and making use of Equations [10.21] and [10.22], we obtain:

$$\left[\delta_* + \frac{1 - n - i\omega_*}{(n + P)^2}\right]\left(\frac{d\omega_*}{dq}\right)_{q=0} + \omega_*\left(\frac{d\delta_*}{dq}\right)_{q=0} = \frac{2}{n + P}\{\pm K\sin\omega_* + i[H + K \mp K(1 - \cos\omega_*)]\}$$

Equating the imaginary part of the two sides, we have:

$$\left(\frac{d\omega_*}{dq}\right)_{q=0} = -\frac{2(n + P)}{\omega_*}\Big[H + K \mp K(1 - \cos\omega_*)\Big] \quad [10.23]$$

Equating the real parts and making use of Equation [10.23] we deduce:

$$\left(\frac{d\delta_*}{dq}\right)_{q=0} = \frac{2(n + P)}{\omega_*^2}\left(\delta_* + \frac{1 - n}{(n + P)^2}\right)\Big[H + K \mp K(1 - \cos\omega_*)\Big] \pm \frac{2K}{n + P}\,\frac{\sin\omega_*}{\omega_*} \quad [10.24]$$

In both Equations [10.23] and [10.24] ω_* and δ_* have the values of Equation [10.22]. We observe first that the quantity in square brackets is always $> H - K$. Now if the values of H, Equation [10.14], and K, Equation [10.1], are computed for various combinations of propellants at the mixture ratio of maximum specific impulse, it is seen that generally H, K are positive and $H > K$. Thus the quantity in square brackets is positive, and $(d\omega_*/dq)_{q=0} < 0$. A few numerical computations show that also the right-hand side of Equation [10.24] is generally positive (except for very large values of P and ω_*, which are improbable); therefore, δ_* increases with q, and Equation [10.17] shows that with short lines (our present assumption) it is possible to obtain a certain improvement of stability by decreasing P_o and increasing P_f, that is, increasing Δp_o and decreasing Δp_f. We observe that this conclusion holds for both assumptions (a) and (b); which seems to give a certain generality to this result.

Next we suppose that J is very large, so that $J\omega \gg 1$. Equation [10.18] is reduced in this case to:

$$[(1 - n + i\omega_*)e^{i\omega_*\delta_*} + n]J\omega_* = iP\,\{1 + 2(j - q)\,[H + K \mp K(1 - e^{-i\omega_*})]\} \quad [10.25]$$

where, being interested in the behavior near $j = q = 0$, we have neglected j^2 and jq with respect to unity (which is rigorously justified in the computation of the derivatives at $j = q = 0$). We see therefore that near $j = q = 0$, the variations of ω_* and δ_* depend only on the quantity $j - q$. When $j = q$, with j and q not too large, the solution of Equation [10.25] is the same as for $j = q = 0$, and is close to the solution, Equation [5.5], of Equation [5.2] with $\alpha = i\omega_*$; which corresponds exactly to $J = \infty$. We can now differentiate Equation [10.25] with respect to $j - q$ at $j - q = 0$. Performing the differentiation, and making use of what gives Equation [10.25] at $j = q$, we can put the result under the form:

$$\left[e^{i\omega_*\delta_*} - n\delta_* + \frac{P}{J\omega_*^2}(1 + i\omega_*\delta_*)\right]\frac{1}{\omega_*}\left[\frac{d\omega_*}{d(j - q)}\right]_{j=q} - \left(n\delta_* - \frac{P}{J\omega_*^2}\cdot i\omega_*\delta_*\right)\frac{1}{\delta_*}\left[\frac{d\delta_*}{d(j - q)}\right]_{j=q} = 2\frac{P}{J\omega_*^2}\Big[H + K \mp K(1 - e^{-i\omega_*})\Big]$$

from which two real equations can be written with the two derivatives as unknowns. Solving for the derivative of δ_* we obtain:

$$\frac{1}{\delta_*}\left[\frac{d\delta_*}{d(j - q)}\right]_{j=q} = -\frac{2P}{J\omega_*^2}\,\frac{\left(\sin\omega_*\delta_* + \frac{P}{J\omega_*^2}\cdot\omega_*\delta_*\right)[H + K \mp K(1 - \cos\omega_*)] \pm \left(\cos\omega_*\delta_* - n\delta_* + \frac{P}{J\omega_*^2}\right)K\sin\omega_*}{n\delta_*\sin\omega_*\delta_* + \frac{P}{J\omega_*^2}\cdot\omega_*\delta_*\left(\cos\omega_*\delta_* + \frac{P}{J\omega_*^2}\right)}$$

Introducing here some representative values of the various quantities and remembering that the quantity in square brackets in the right-hand side is generally positive, it is possible to see that the derivative is negative in most of the cases, and therefore an improvement of stability can be obtained by making $q - j$ positive;

that is, by increasing Δp_o, J_o, and decreasing Δp_f, J_f.

These conclusions are of course only qualitative; a more complete and quantitative discussion would be too long and involved for the present paper; and, perhaps, would not be justified by the roughness of our combustion assumptions.

11 Temperature Nonuniformity Due to Pressure Oscillations

So far we have always assumed that the temperature of the gases in the chamber is uniform [monopropellant case; bipropellant case, assumption (a)]; or is a function of the mixture ratio [bipropellant case, assumption (b)]. Now, if it is true that at the instant of combustion every portion of the gases is produced at the same temperature, or at a temperature depending on the mixture ratio alone (neglecting secondary effects of the pressure on combustion), it is also true that after they have been produced the gases will undergo changes in temperature when the pressure changes; and that in the hypothesis that mixing and dissipations are negligible, these changes will be practically isentropic. Hence, in the case of a monopropellant, if we consider a particle at a reduced time ϵ after the instant of combustion, we can relate the fractional variation of temperature with respect to the constant combustion temperature with the fractional variation of pressure with respect to the pressure at the instant of combustion through the equation

$$\frac{T_g - \bar{T}_g}{\bar{T}_g} \cong \frac{\gamma - 1}{\gamma}\,\frac{p - p^{(\epsilon)}}{p^{(\epsilon)}} \cong \frac{\gamma - 1}{\gamma}(\varphi - \varphi^{(\epsilon)}) \quad [11.1]$$

where the fractional variations have been taken as being small. About ϵ we make the same rough but simple assumption we have used in Section 10: that the combustion takes place at the injection end, and that the residence time is θ_g for all the particles; so that at the exhaust end $\epsilon = 1$. For a certain element of volume dV of the combustion chamber we can find the time dt necessary to the gases to cross dV from Equation [10.10]:

$$dt = dV\,\frac{\theta_g}{V}$$

The total time from the combustion instant up to this element of volume is equal to $\int dt = \theta_g\epsilon$; therefore

$$\frac{dV}{V} = d\epsilon \quad [11.2]$$

From the first Equation [10.2] and Equation [11.1], we have, therefore, in the monopropellant case

$$\mu_\epsilon = \varphi + \frac{\gamma - 1}{2\gamma}(\varphi - \varphi^{(1)}) \quad [11.3]$$

and from the second Equation [10.2], or Equations [10.8] and [10.9], and Equations [11.1], [11.2]:

$$\frac{M_g}{\bar{M}_g} = 1 + \varphi + y;$$

$$1 + y = \frac{1}{V}\int_i^\epsilon dV\left[1 - \frac{\gamma - 1}{\gamma}(\varphi - \varphi^{(\epsilon)})\right] = 1 - \frac{\gamma - 1}{\gamma}\int_0^1 (\varphi - \varphi^{(\epsilon)})\,d\epsilon$$

The integral can be easily evaluated if φ varies as exp (αz); the result is

$$y = -\frac{\gamma - 1}{\gamma}\left[1 + \frac{1}{\alpha}(e^{-\alpha} - 1)\right]\varphi(z)$$

and therefore

$$\frac{dy}{dz} = -\frac{\gamma - 1}{\gamma}\left(\frac{d\varphi}{dz} + \varphi^{(1)} - \varphi\right) \quad [11.4]$$

can be replaced in the second Equation [10.9] to find the value of the first term of Equation [10.6].

In the bipropellant case the fractional variation due to changes in mixture ratio is to be taken into account by adding in Equations [11.3] and [11.4] the corresponding terms of Equations [10.5] and [10.11].

We will discuss here only the effects of the additional terms in the simplest case of intrinsic instability, with a monopropellant. Instead of Equation [4.8] with $\mu^{(\delta)} = 0$, we have now the equation deduced from Equation [10.6] with the previously derived values for the first two terms and with the same μ_b as in Section 4:

$$\frac{1}{\gamma}\frac{d\varphi}{dz} + \frac{\gamma - 1}{2\gamma}(\varphi - \varphi^{(1)}) + \varphi = n(\varphi - \varphi^{(\delta)});$$

taking φ varying as exp $(i\omega_*)$ we have the complex equation:

$$1 - n + \frac{i\omega_*}{\gamma} + \frac{\gamma - 1}{2\gamma}(1 - e^{-i\omega_*}) + ne^{-i\omega_*\delta_*} = 0$$

Equating the moduli and the arguments we find the two real equations in ω_*, δ_*:

$$\left.\begin{aligned} &\left[1 - n + \frac{\gamma - 1}{2\gamma}(1 - \cos\omega_*)\right]^2 + \left(\frac{\omega_*}{\gamma} + \frac{\gamma - 1}{2\gamma}\sin\omega_*\right)^2 = n^2 \\ &\omega_*\delta_* = \pi - \tan^{-1}\frac{\dfrac{\omega_*}{\gamma} + \dfrac{\gamma - 1}{2\gamma}\sin\omega_*}{1 - n + \dfrac{\gamma - 1}{2\gamma}(1 - \cos\omega_*)} \end{aligned}\right\} \quad [11.5]$$

From the first Equation [11.5] we see that when $n = 0.5$, $\omega_* = 0$ satisfies the equation, exactly as in the case discussed in Section 5. When $n < 0.5$, ω_* is imaginary and the system is always stable. If n is only slightly larger than 0.5, ω_* is real but small. We can therefore take approximately $\sin\omega_* \cong \omega_*$ and $\cos\omega_* \cong 1$; and the solution of Equation [11.5] is immediately found to be

$$\omega_* \cong \frac{2\gamma}{\gamma + 1}\sqrt{2n - 1};$$

$$\delta_* = \frac{\gamma + 1}{2\gamma}\,\frac{1}{\sqrt{2n - 1}}\left(\pi - \tan^{-1}\frac{\sqrt{2n - 1}}{n - 1}\right)$$

which is very similar to Equation [5.5] except for a constant numerical factor. When $\gamma = 1.3$, $(\gamma + 1)/2\gamma = 0.885$; so that the critical δ_* is reduced by about 11.5%, and ω_* increased of the same amount with respect to the values of Section 4 and Fig. 2. If, on the other hand, we take $n = 1$, the Equations [11.5] for $\gamma = 1.3$ have the solution $\omega_* = 1.175$; $\delta_* = 0.855\ \pi/2$, instead of the values $\omega_* = 1$; $\delta_* = \pi/2$ of the corresponding case of Section 5. The critical δ_* is there-

fore decreased also in this case by about 14.5%. We see therefore that for values of n between 0.5 and 1 the stability decreases due to the effect of the temperature fluctuations. The percentual decrease of δ_* is contained between 11.5% and 14.5% with $\gamma = 1.3$, and is reduced with decreasing γ (when $\gamma = 1$ there is no decrease at all). Of course the numerical results are only good with the assumptions made. Qualitatively, however, it seems that even with a more realistic behavior of the combustion we should still expect a decrease of stability, but of even smaller importance.

12 Nonuniformity of the Pressure and Possibility of High-Frequency, Self-Excited Organ-Pipe Oscillations

In all the preceding Sections we have assumed that the pressure waves are propagated so fast through the combustion chamber that practically at every instant the pressure is uniform at all points. This is why, except for secondary effects studied in the last two Sections, no assumption was necessary on the spatial distribution of the combustion process. However, if we want to analyze the effects of wave propagation and nonuniform distribution of the pressure, we have also to know how the combustion is distributed in the combustion chamber. Assuming, for simplicity, a cylindrical combustion chamber, we could suppose in steady operation a given law of gas production along the length of the chamber and try to solve the problem of the stability of small perturbations. But the mathematical treatment of this problem is not easy; so that we have preferred to confine our analysis to the simplest case, already considered in the last two Sections, when practically all of the combustion takes place at the injector end, and the rest of the combustion chamber can be considered just as a constant section duct, where in steady operation the values of pressure, temperature, and velocity of the gases are uniform. Of course the quantitative results obtained with this simplified model will not correspond exactly to any real case, except for very large values of L^*; nevertheless, they will give an idea of the qualitative behavior of real cases.

As there is no gas production along the chamber, the flow in the chamber is controlled by the ordinary equations of isentropic nonsteady, one-dimensional motion of gases:

$$\rho_g(u_t + uu_x) = -p_x; \quad (\rho_g)_t + (\rho_g u)_x = 0; \quad p\rho_g^{-\gamma} = \text{const.} \qquad [12.1]$$

where the subscripts t and x mean partial differentiation with respect to the corresponding variables. In the hypothesis of small perturbations, introducing the fractional variations around the steady state values $\bar{u}$, $\bar{\rho}_g$, $\bar{p}$:

$$u = \bar{u}(1 + \nu); \quad \rho_g = \bar{\rho}_g(1 + \sigma); \quad p = \bar{p}(1 + \varphi);$$

introducing also the sound velocity and its fractional variations

$$c^2 = \frac{dp}{d\rho_g} = \gamma\frac{p}{\rho_g}; \quad c = \bar{c}\left(1 + \frac{\varphi - \sigma}{2}\right) \qquad [12.2]$$

we have first from the third Equation [12.1] and from the second Equation [12.2],

$$\varphi = \gamma\sigma; \quad c = \bar{c}\left(1 + \frac{\gamma - 1}{2}\sigma\right) \qquad [12.3]$$

Then from the first and second Equation [12.1], neglecting the terms of higher order in the fractional variation and eliminating p through the first Equation [12.2], we obtain the linear system:

$$\bar{u}\nu_t + \bar{u}^2\nu_x + \bar{c}^2\sigma_x = 0; \quad \sigma_t + \bar{u}\sigma_x + \bar{u}\nu_x = 0 \qquad [12.4]$$

If we try a solution of the form $\sigma = \sigma(\zeta)$; $\nu = \nu(\zeta)$ with $\zeta = t + ax$, we have $\sigma_t = \sigma_x/a = \sigma_\zeta$ and $\nu_t = \nu_x/a = \nu_\zeta$, so that the equations are reduced to:

$$\bar{u}(1 + \bar{u}a)\nu_\zeta + \bar{c}^2a\sigma_\zeta = 0; \quad (1 + \bar{u}a)\sigma_\zeta + \bar{u}a\nu_\zeta = 0 \qquad [12.5]$$

In order to have nonzero solutions for ν_ζ and σ_ζ, the determinant of the coefficients must be zero; that is:

$$\bar{u}[(1 + \bar{u}a)^2 - \bar{c}^2a^2] = 0$$

Therefore there are two values of a;

$$a_1 = \frac{1}{\bar{c} - \bar{u}}; \quad a_2 = -\frac{1}{\bar{c} + \bar{u}} \qquad [12.6]$$

So that the general solution of Equation [12.4] is of the form:

$$\sigma = \sigma_1(t + a_1x) + \sigma_2(t + a_2x); \quad \nu = \nu_1(t + a_1x) + \nu_2(t + a_2x) \qquad [12.7]$$

Replacing in Equation [12.5] and integrating with the condition $\sigma = 0$ when $\nu = 0$ (steady-state condition), we find that the four arbitrary functions of Equation [12.7] are bound by the relations:

$$\sigma_1 = -\frac{\bar{u}}{\bar{c}}\nu_1 = -\bar{M}\nu_1; \quad \sigma_2 = \frac{\bar{u}}{\bar{c}}\nu_2 = \bar{M}\nu_2$$

where $\bar{M}$ is the Mach number in steady-state flow. The general solution of Equation [12.4] is therefore:

$$\nu = \nu_1(t + a_1x) + \nu_2(t + a_2x); \quad \sigma = \bar{M}[-\nu_1(t + a_1x) + \nu_2(t + a_2x)] \qquad [12.8]$$

We have now to write the boundary conditions. At the exhaust end we have a nozzle with a sonic throat. If the subsonic portion of the nozzle is sufficiently short the wave propagation in this portion takes a very short time, and we can approximately represent the flow conditions with the steady-state conditions. In this case the Mach number at the exhaust end of the chamber will stay unchanged at all instants, since it is determined only by the ratio of the cross section of the chamber to the area of the throat. Then the boundary condition at the exhaust end is $M = \bar{M}$ at $x = L$. But from Equation [12.3]

$$M = \frac{u}{c} = \bar{M}\left(1 + \nu - \frac{\gamma - 1}{2}\sigma\right);$$

Therefore $\nu = (\gamma - 1)\sigma/2$ at $x = L$; that is, by Equation [12.8],

$$\left(1 + \frac{\gamma - 1}{2}\bar{M}\right)\nu_1(t + a_1L) + \left(1 - \frac{\gamma - 1}{2}\bar{M}\right)\nu_2(t + a_2L) = 0 \quad [12.9]$$

At the injector end we have all of the gas production under variable pressure condition, the pressure variation being given by the first equation [12.3], with σ computed from [12.8] at $x = 0$:

$$\varphi(t) = \gamma\bar{M}[-\nu_1(t) + \nu_2(t)]$$

The mass flow at $x = 0$ is at every instant equal to the burning rate. In steady state these are equal to the injection rate:

$$\bar{\dot{m}}_b = \bar{\dot{m}}_i = \bar{\rho}_g\bar{u}A_c$$

In nonsteady state we have from Equation [4.3] supposing a constant injection rate:

$$\dot{m}_b = \bar{\dot{m}}_i\left(1 - \frac{d\tau}{dt}\right) = \bar{\rho}_g\bar{u}A_c\,(1 + \nu + \sigma) \text{ at } x = 0$$

Therefore, introducing the value of $d\tau/dt$ from Equation [4.8], we find

$$\nu + \sigma = n(\varphi - \varphi^{(\bar{\tau})}) = n\gamma(\sigma - \sigma^{(\bar{\tau})}) \text{ at } x =$$

To simplify the following analysis we will now assume $n\gamma = 1$, so that $n = 1/\gamma$ has a value between 0.5 and 1. The case of general n can be easily developed on the same line of the following development. In this case the condition at the injector end is simplified to:

$$\nu + \sigma^{(\bar{\tau})} = 0 \text{ at } x = 0$$

that is, introducing Equations [12.8]:

$$\nu_1(t) - \bar{M}\nu_1(t - \bar{\tau}) + \nu_2(t) + \bar{M}\nu_2(t - \bar{\tau}) = 0 \quad [12.10]$$

More than in the general solution of our problem we are interested in the stability condition. Following the same technique used in the preceding sections, let us try a solution of the kind:

$$\nu_1 = C_1e^{\alpha(t + a_1x)};\quad \nu_2 = C_2e^{\alpha(t + a_2x)} \quad [12.11]$$

when α, C_1, C_2, are (generally) complex constants.

Introducing Equation [12.11] in Equation [12.9] and writing

$$B = \frac{1 + \dfrac{\gamma - 1}{2}\bar{M}}{1 - \dfrac{\gamma - 1}{2}\bar{M}} \quad [12.12]$$

we find, eliminating the common factor exp (αt):

$$BC_1e^{\alpha a_1L} + C_2e^{\alpha a_2L} = 0$$

that is

$$\frac{C_2}{C_1} = -Be^{\alpha\Theta} \quad [12.13]$$

where from Equation [12.6]

$$\Theta = (a_1 - a_2)L = \frac{L}{\bar{c} - \bar{u}} + \frac{L}{\bar{c} + \bar{u}} = \frac{2\bar{c}L}{\bar{c}^2 - \bar{u}^2} \quad [12.14]$$

is a characteristic time of the combustion chamber, representing the total time of propagation of a pressure wave from the injector end to the exhaust end and back to the injector end. Introducing now the tentative solution, Equation [12.11], into Equation [12.10] and eliminating the common factor exp (αt), we find

$$C_1 + C_2 - \bar{M}e^{-\alpha\bar{\tau}}(C_1 - C_2) = 0$$

that is, dividing by C_1 and using Equation [12.13]

$$1 - Be^{\alpha\Theta} = \bar{M}e^{-\alpha\bar{\tau}}(1 + Be^{\alpha\Theta})$$

Introducing $\alpha = \Lambda + i\Omega$, and solving once for $\bar{M}$ exp $(-\alpha\bar{\tau})$ and once for B exp$(\alpha\Theta)$, this equation can be put in the two equivalent forms:

$$\left.\begin{aligned} \bar{M}e^{-(\Lambda\bar{\tau} + i\Omega\bar{\tau})} &= -\frac{Be^{\alpha\Theta} - 1}{Be^{\alpha\Theta} + 1} = \\ &-\frac{Be^{\Lambda\Theta}\cos\Omega\Theta - 1 + iBe^{\Lambda\Theta}\sin\Omega\Theta}{Be^{\Lambda\Theta}\cos\Omega\Theta + 1 + iBe^{\Lambda\Theta}\sin\Omega\Theta} \end{aligned}\right\} \quad [12.15]$$

$$Be^{\Lambda\Theta + i\Omega\Theta} = \frac{1 - \bar{M}e^{-\alpha\bar{\tau}}}{1 + \bar{M}e^{-\alpha\bar{\tau}}} = \frac{1 - \bar{M}e^{-\Lambda\bar{\tau}}\cos\Omega\bar{\tau} + i\bar{M}e^{-\Lambda\bar{\tau}}\sin\Omega\bar{\tau}}{1 + \bar{M}e^{-\Lambda\bar{\tau}}\cos\Omega\bar{\tau} - i\bar{M}e^{-\Lambda\bar{\tau}}\sin\Omega\bar{\tau}}$$

Equating the squares of the moduli of the two equations we obtain:

$$\bar{M}e^{2-2\Lambda\bar{\tau}} = \frac{B^2e^{2\Lambda\Theta} + 1 - 2Be^{\Lambda\Theta}\cos\Omega\Theta}{B^2e^{2\Lambda\Theta} + 1 + 2Be^{\Lambda\Theta}\cos\Omega\Theta}$$

$$B^2e^{2\Lambda\Theta} = \frac{1 + \bar{M}^2e^{-2\Lambda\bar{\tau}} - 2\bar{M}e^{-\Lambda\bar{\tau}}\cos\Omega\bar{\tau}}{1 + \bar{M}^2e^{-2\Lambda\bar{\tau}} + 2\bar{M}e^{-\Lambda\bar{\tau}}\cos\Omega\bar{\tau}}$$

and finally solving for cos $(\Omega\Theta)$ and cos $(\Omega\bar{\tau})$

$$\left.\begin{aligned} \cos\Omega\Theta &= \frac{B^2e^{2\Lambda\Theta} + 1}{2Be^{\Lambda\Theta}}\cdot\frac{1 - \bar{M}^2e^{-2\Lambda\bar{\tau}}}{1 + \bar{M}^2e^{-2\Lambda\bar{\tau}}} \\ \cos\Omega\bar{\tau} &= -\frac{1 + \bar{M}^2e^{-2\Lambda\bar{\tau}}}{2\bar{M}e^{-\Lambda\bar{\tau}}}\cdot\frac{B^2e^{2\Lambda\Theta} - 1}{B^2e^{2\Lambda\Theta} + 1} \end{aligned}\right\} \quad [12.16]$$

We see that for given $\bar{M}$ (and therefore B, Equation [12.12]) and given Θ, $\bar{\tau}$, these two equations are sufficient to find the values of Ω and Λ (therefore α) for which the Equation [12.11] represents a solution of our problem. We see that if Ω satisfies our system, $-\Omega$ satisfies it too; therefore by proper combinations of solutions with complex conjugate α-values, real oscillatory solutions can be found, stable or unstable following if Λ is negative or positive. Only positive values of Ω need then to be considered. The indetermination in the signs of $\Omega\Theta$ and $\Omega\bar{\tau}$ when the cosines are given can be overcome by equating the imaginary parts of, for instance, Equation [12.15]; we find

$$\bar{M}e^{-\Lambda\bar{\tau}}\sin\Omega\bar{\tau} = \frac{2Be^{\Lambda\Theta}\sin\Omega\Theta}{1 + B^2e^{2\Lambda\Theta} + 2Be^{\Lambda\Theta}\cos\Omega\Theta} \quad [12.17]$$

and, as the quantity below the line is always positive, we see that sin $(\Omega\bar{\tau})$ and sin $(\Omega\Theta)$ must always have the same sign.

The values of $\Omega\bar{\tau}$ and $\Omega\Theta$ can be connected to quantities having a more direct physical meaning. In fact, every real solution of our problem can be expressed as a combination of nonoscillatory functions of the time, times sinusoidal functions of the two quantities $\Omega(t + a_1x)$, $\Omega(t + a_2x)$. At a given location x the result will be a sinusoidal function of Ωt with a certain phase angle, and the local period of oscillation is given by $\Omega T = 2\pi$; therefore

$$\frac{\Omega\bar{\tau}}{2\pi} = \frac{\bar{\tau}}{T} \quad [12.18]$$

represents how many periods of local oscillation are contained in the time lag. On the other hand, for a fixed value of t a combination of two sinusoidal functions of $\Omega a_1 x$ and $\Omega a_2 x$ will represent the spatial distribution of the solution at the time t. If $\bar{M}$ is small, and therefore u can be neglected with respect to $\bar{c}$, we have from Equation [12.6] $a_1 \cong -a_2 \cong -1/\bar{c}$, and the spatial distribution is represented by a single sinusoidal function of $\Omega x/\bar{c}$. The half wave length X of the distribution is then given by $\Omega X/\bar{c} = \pi$. Therefore, as in the present assumption $\Theta \cong 2L/\bar{c}$, we deduce that the quantity

$$\frac{\Omega\Theta}{2\pi} = \frac{L}{X} \quad [12.19]$$

represents approximately the number of half wave lengths contained in the length of the combustion chamber.

Let us now, as in the previous Section, determine the critical values of the time lag $\bar{\tau}_*$ corresponding to assigned values of $\bar{M}$ and Θ, and to a certain value Ω_*, also to be determined. We obtain the critical condition by putting $\Lambda = 0$ in Equations [12.16], so that

$$\cos \Omega_*\Theta = \frac{B^2 + 1}{2B} \cdot \frac{1 - \bar{M}^2}{1 + \bar{M}^2};$$

$$\cos \Omega_*\bar{\tau}_* = -\frac{1 + \bar{M}^2}{2\bar{M}} \cdot \frac{B^2 - 1}{B^2 + 1} \quad [12.20]$$

The first Equation [12.20] gives the values of Ω_* and then the second gives the value of $\bar{\tau}_*$. From Equation [12.12] we have

$$\frac{B^2 + 1}{2B} = \frac{1 + \left(\frac{\gamma - 1}{2}\bar{M}\right)^2}{1 - \left(\frac{\gamma - 1}{2}\bar{M}\right)^2} \cong 1;$$

$$\frac{B^2 - 1}{B^2 + 1} = \frac{(\gamma - 1)\bar{M}}{1 + \left(\frac{\gamma - 1}{2}\bar{M}\right)^2} \cong (\gamma - 1)M$$

the approximate values being very accurate even for large values of $\bar{M}$, due to the factor $(\gamma - 1)/2$. Therefore

$$\cos \Omega_*\Theta \cong \frac{1 - \bar{M}^2}{1 + \bar{M}^2}, \text{ or } \sin \Omega_*\Theta \cong \pm \frac{2\bar{M}}{1 + \bar{M}^2};$$

$$\cos \Omega_*\bar{\tau}_* \cong -\frac{\gamma - 1}{2}(1 + \bar{M}^2)$$

Solving these trigonometrical equations we find

$$\Omega_*\Theta \cong 2k\pi \pm \sin^{-1}\frac{2\bar{M}}{1 + M^2}; \quad \Omega_*\bar{\tau}_* \cong 2h\pi \pm \left[\frac{\pi}{2} + \sin^{-1}\frac{\gamma - 1}{2}(1 + \bar{M}^2)\right] \quad [12.21]$$

h and k being zero or positive integers. When h or k is zero, only the upper sign can be used in Equation [12.21], since Ω_* is by definition a positive quantity.

The second Equation [12.21] can also be written

$$\Omega_*\bar{\tau}_* \cong (2h + 1)\pi \mp \left[\frac{\pi}{2} - \sin^{-1}\frac{\gamma - 1}{2}(1 + \bar{M}^2)\right] \quad [12.22]$$

with $h = 0$ or a positive integer. In this equation both signs are possible when $h = 0$.

The above expressions are sufficiently accurate, under the assumptions of this section, up to $\bar{M} = 1$ (throatless motor). However the combustion assumptions certainly become less and less satisfactory for increasing values of $\bar{M}$. If $\bar{M}$ is sufficiently small the first Equation [12.21] and Equation [12.22] can be written:

$$\left.\begin{aligned} &\Omega_*\Theta \cong 2k\pi \pm 2\bar{M}; && (k = 0, 1, \ldots) \\ &\Omega_*\bar{\tau}_* \cong (2h + 1)\pi \mp \left(\frac{\pi}{2} - \frac{\gamma - 1}{2}\right); && (h = 0, 1, \ldots) \end{aligned}\right\} \quad [12.23]$$

where we have also made the approximation

$$\sin[(\gamma - 1)/2] \cong (\gamma - 1)/2.$$

The choice of the signs in Equations [12.21] to [12.23] is restricted by the conditions derived from Equation [12.17], namely that $\sin(\Omega\Theta)$ and $\sin(\Omega\bar{\tau})$ have always the same sign. This means that in the equations just written we have to select both upper signs, or both lower signs.

Equations [12.23], or the corresponding ones more accurate, define certain critical values of Ω_* and $\bar{\tau}_*$; at these values Λ is zero and we have transition from stable to unstable, or vice versa. Without entering in a detailed discussion of Equation [12.16], it is possible to show that for given values of h, k, Λ is positive, and therefore the conditions are unstable, whenever the inequality

$$\frac{(2h + 1)\pi - \left(\frac{\pi}{2} - \frac{\gamma - 1}{2}\right)}{2k\pi + 2\bar{M}} < \frac{\bar{\tau}}{\Theta} < \frac{(2h + 1)\pi + \left(\frac{\pi}{2} - \frac{\gamma - 1}{2}\right)}{2k\pi - 2\bar{M}} \quad [12.24]$$

corresponding to the approximation of Equations [12.23], or the similar one derived from Equations [12.21], [12.22], is satisfied; in the opposite case the Λ value of the corresponding oscillation mode is negative and the solution is stable. The maximum value of Λ (and the maximum amplification rate for the corresponding oscillation mode) is obtained at

$$\frac{\bar{\tau}}{\Theta} = \frac{2h + 1}{2k} = \frac{h}{k} + \frac{1}{2k}$$

which therefore represents the most dangerous value of the time lag for the corresponding oscillation mode.

The meaning of h, k, is directly obtained comparing Equations [12.23] with Equations [12.18], [12.19]. We see that h represents approximately the number of periods of oscillation contained in the time lag at a given location; and k the number of half-wave lengths contained in the length of the combustion chamber

If we take $h = 0$, $k = 0$, we obtain the fundamental mode of oscillation. In this case the right-hand side of Equation [12.24] is negative and has no physical meaning; and therefore we conclude, considering only the left-hand side, that the fundamental mode is stable or unstable following if:

$$\frac{\bar{\tau}}{\Theta} \lessgtr \frac{\frac{\pi}{2} + \frac{\gamma - 1}{2}}{2\bar{M}} \qquad [12.25]$$

It is interesting to compare this result with the result of Section 5. From the definition of Θ, Equation [12.14], and from the definition of the gas residence time which, under the present assumption of constant section and velocity throughout the combustion chamber, is equivalent to $\theta_g = L/\bar{u}$, we obtain

$$\Theta = \frac{2\bar{c}\bar{u}}{\bar{c}^2 - \bar{u}^2}\,\theta_g = \frac{2\bar{M}}{1 - \bar{M}^2}\,\theta_g$$

or, for small $\bar{M}$, $\Theta = 2\bar{M}\theta_g$. Hence Equation [12.25] becomes

$$\frac{\bar{\tau}}{\theta_g} \lessgtr \frac{\pi}{2} + \frac{\gamma - 1}{2} \qquad [12.26]$$

which shows a value of the critical time lag very close to the one computed with $n = 1$ at Section 5 with the assumption of uniform pressure. The reason is that when $k = 0$ and $\bar{M}$ is small, Equations [12.23] and [12.19] show that $L/X = \bar{M}/2\pi$ is a small number, so that the length of the combustion chamber is a small fraction of a half-wave length and at every instant the pressure is nearly constant in the combustion chamber. In this case the gases can be assumed to oscillate as a whole, as we have assumed in the previous sections. This is also true if we take $k = 0$, $h \neq 0$. Again the corresponding mode of oscillation is stable or unstable following if

$$\frac{\bar{\tau}}{\Theta} \lessgtr \frac{2h\pi + \frac{\pi}{2} + \frac{\gamma - 1}{2}}{2M}$$

which shows a value of the critical time lag always larger than the one of Equation [12.25], in agreement with the results of Section 5, concerning the higher frequency modes for which more than one period is contained in the time lag.

Of course if $\bar{M}$ is large, the more exact Equations [12.21], [12.22] have to be used, and the divergence between the results and those obtained with uniform pressure conditions increases. But in this case the combustion assumptions of this section become more questionable.

If $k \neq 0$, and has a fixed value, we again obtain the lowest critical time lag when $h = 0$, and the corresponding mode is unstable when

$$\frac{\frac{\pi}{2} + \frac{\gamma - 1}{2}}{2k\pi + 2\bar{M}} < \frac{\bar{\tau}}{\Theta} < \frac{\frac{3\pi}{2} - \frac{\gamma - 1}{2}}{2k\pi - 2\bar{M}} \qquad [12.27]$$

These are the limits of instability of the lowest frequency mode with k half-wave lengths in the length of the chamber. The corresponding angular frequency satisfies the inequality

$$2k\pi - 2\bar{M} < \Omega_* \,\Theta < 2k\pi + 2\bar{M}$$

and is very close to the one characterizing the corresponding kth harmonic of the organ-pipe oscillations, which is given by $\Omega_k\Theta = 2k\pi$. With $k = 1$ we have the fundamental mode of the organ-pipe oscillations corresponding to the lowest high-frequency mode in our case.

The ratio of the frequency of this mode to the frequency of the fundamental mode of oscillating combustion ($k = 0$) is about $\pi/\bar{M}$ which is generally a number of the order of ten or more.

When τ/Θ is larger than the right-hand side of Equation [12.27], the mode with the given k value and $h = 0$ is again stable; but above a certain value of the time lag the inequality

$$\frac{2\pi + \frac{\pi}{2} + \frac{\gamma - 1}{2}}{2k\pi + 2\bar{M}} < \frac{\bar{\tau}}{\Theta} < \frac{2\pi + \frac{3\pi}{2} - \frac{\gamma - 1}{2}}{2k\pi - 2\bar{M}} \qquad [12.28]$$

will in turn be satisfied, and the mode with the given k, and $h = 1$ becomes unstable. We see now how with increasing $\bar{\tau}$ the different modes corresponding to a given k become successively unstable following Equation [12.24]. If we take into account all of the possible values of h, k we see that for no value of the time lag all the modes are stable; with every value of $\bar{\tau}$, no matter how small, there are unstable modes. However, the smaller $\bar{\tau}$, the higher is the minimum value of k corresponding to an unstable mode.

On a purely qualitative basis it seems now that modes with high values of k are probably not too important, since little deviations from the assumed idealized conditions of combustion will probably kill them off. However, the instability of modes with small values of k ($k = 1$ or 2) can be worse than the one of the fundamental mode, since they are associated with higher frequency local velocity fluctuations (absent in the case of the fundamental mode where the gases oscillate practically as a whole) and these velocity fluctuations may be responsible for large increases in heat transfer. Equation [12.28] shows that one of these modes can be unstable, even when the fundamental low-frequency mode is stable, and that, for this kind of instability, stability can be obtained both increasing or decreasing $\bar{\tau}/\Theta$, that is, working on the injector system to modify $\bar{\tau}$ or on the combustion chamber to change Θ. However, if the fundamental mode is unstable, the only possibility to reach stability is afforded by a decrease of $\bar{\tau}/\Theta$.

We conclude this section observing that the quantitative results obtained hold only under the special assumptions made on the way the combustion takes place, and only for $n = 1/\gamma$. However, qualitatively the results are simple enough to allow their generalization; we have found that, in the same way as in previous sections for the combustion chamber as a whole, a local interference exists between the physicochemical processes that lead to the transformation of the propellants into hot gases and the pressure variation; and that under proper circumstances this interference may give rise to different modes of self-excited oscillations and rough combustion. We also see that a fundamental mode of oscillation, where the pressure oscillates nearly simultaneously at all points in the combustion chamber, is excited when the time lag is above a certain value, which can be determined from the analysis

of the preceding section: and that frequencies close to the natural modes of organ-pipe oscillation are excited when the time lag is contained in certain ranges, which in their totality cover all the possible values of the time lag, so that no value of the time lag can be found for which the combustion is rigorously stable. However, if the time lag is sufficiently small the corresponding unstable modes are not likely to be very important. It seems reasonable now to generalize these results to the case when the combustion process is diffused in the totality of the combustion chamber and when the shape of the latter is such that more complicated modes of oscillation have to be taken into account, the generalization being obtained by simply dropping the words "organ-pipe" specifying the kind of natural modes of oscillation.[4] However, the quantitative determination of the actual dangerous ranges of the time lag would require a much more elaborate analysis than the one performed in this section.

Finally, observe that the analysis performed could be extended to other values of n, with the probable result that this kind of instability is only possible if n is in excess of a certain minimum value.

13 Summary of Conclusions

1 The physicochemical processes responsible for the transformation of the propellants into hot gases at the end of the time lag are affected by the pressure and by the pressure variation. Although very little is known today on these processes, a reasonable empirical relation is obtained with the assumption that the rate at which they take place is in average proportional to a power of the local and instantaneous pressure. Two empirical formulas are then suggested connecting the time lag with pressure: one, the simpler, contains two arbitrary constants; the second, which is more general, three. In the analysis the first one has been selected.

2 Neglecting the pressure nonuniformity in the chamber, as it is allowed for the low-frequency oscillations, it is shown that self-excited oscillations can be generated even with a rigorously constant injection rate. This so-called intrinsic instability can only exist when the pressure exponent of the empirical formula for the rate of the processes during the time lag is larger than 0.5, and when the time lag is larger than a certain critical value, function of the parameters of the combustion chamber.

3 If the feeding system is sensitive to variations in chamber pressure, the stability is generally decreased with respect to the case of constant injection rate.[5] Therefore an intrinsically unstable rocket motor cannot generally be made stable by changes in the feeding system. The quantitative determination of the critical time-lag and of the critical frequency has been made for two general types of feeding systems, one with a constant rate supply, the other with a constant pressure supply, and both with a concentrated elasticity in the feeding line. The results are described in Figs. 8, 9, and 10 and discussed in Sections 7 and 8. It is important to observe that the frequency determined in this way can be applied only to the critical condition of incipient instability, and therefore cannot be compared with actual values observed in fully unstable operation.

4 If the time lag, instead of being assumed uniform, is different for different portions of the injected propellant, the resulting change in the critical conditions shows an improvement of the intrinsic stability. An analogous improvement is likely to be found for more general types of feeding systems. It is interesting to observe that even with large degrees of nonuniformity of the time lag, the results are still close to the ones obtained with the assumption of a uniform average value of the time lag.

5 The equations of a bipropellant system with constant pressure supply have been derived. A general discussion of these equations is difficult due to the increased number of parameters involved. Two interesting results are: (a) If the pressure drop of the two feeding systems are the same and the product of the length and the mass velocity is the same for both feeding lines, the equations become the same as for a monopropellant system, so that all the corresponding results can be applied; (b) starting from this condition and supposing the mixture ratio is adjusted for maximum thrust an improvement of stability is generally obtained when the pressure drop of the oxidizer and a certain parameter of the corresponding line are both increased with respect to those of the fuel, the mean values being kept constant.

6 The effect of the pressure oscillations on the temperature of the gases after they have been formed, which had been neglected so far, has been analyzed. The result is a slight decrease of intrinsic stability.

7 Self-excited high-frequency oscillations with frequencies close to the natural modes of oscillation are possible, as a result of local interaction between different modes of pressure oscillation and combustion. Quantitatively the only case examined is the one of organ-pipe oscillations with combustion concentrated at the injection end of the combustion chamber. In this case, every mode of oscillation becomes unstable in a certain range of time lags; and for every possible value of the time lag there are unstable modes, though the order of unstable modes increases with decreasing time lag.

[4] There is a certain degree of indetermination in the actual computation of the frequency of these natural modes, since the conditions of the gases in the combustion chamber are not well determined, due to the presence of unburned and not completely burned gases. For the time being it seems reasonable for this purpose to assume that the chamber is filled only with gases in the conditions following a complete combustion.

[5] It has been seen in Section 7 that the opposite is exceptionally true for a constant rate supply and a small degree of elasticity in the lines. However, even in this case the improvement of stability is small.

Rocket Propulsion Progress: A Literature Survey

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THIS article presents a summary of the technical progress achieved in the field of rocket propulsion as reflected in the published literature, and is also a guide to outstanding articles in the field. Some 700 articles and about 25 books have appeared on various phases of the subject in the past 30 years. Since the majority of these publications were written only in the past few years, it is apparent that the progress in this field has indeed been rapid.

The scope of this summary necessarily has been limited so that reference is made here to some 230 of the outstanding unclassified books and papers. Development activities covered by military security regulations are not mentioned, of course, but the general trends indicated in this report are nevertheless valid. Often several publications deal with the same detail subject and, in these cases, reference is made only to one or two typical articles rather than to all of them. The author would appreciate learning of other articles concerning related subjects not specifically mentioned here.

The literature survey permits comparison of early articles with recent papers and thus furnishes an evaluation of the progress. In general, the advancement of the science has been most satisfactory. The transition from the enthusiastic part-time investigator and the rocket amateur to the trained and specialized propulsion engineer and researcher has been achieved. The basic problems have been solved or partly solved and the rocket is on its way to becoming a reliable, versatile, and proved means of locomotion. Nevertheless, several difficult phenomena are yet to be clarified, and additional production and field experience are badly needed.

1 General Advances and Theory

1.1 *General*

The over-all advancement of the rocket propulsion field has now been well documented by a number of general books and articles. Some of these books are semitechnical and suitable as first readers (1.101, 1.102).[2] Others give a history of rocket development (1.103), a technical introduction to the subject suitable for use as a college text (1.104, 1.105), or a record of early classical advances (1.106, 1.107).

As in other new fields, there is a large number of general articles that explain in more or less technical language the general principles, types, and applications of rocket propulsion. Only some of these are referenced here (1.108–1.113). They form a good introduction to the field and aid in the training of specialized personnel.

1.2 *Thermodynamics*

The theoretical relations of gas flow through a nozzle have generally been verified, and simple one-dimensional theory can now be used for rocket nozzle and chamber flow calculations (1.201). The curve-chart form suggested by Malina has been useful in many nozzle calculations (1.202). Investigations have shown that the simple theory is not sufficient in overexpanding nozzles (1.203, 1.204): the nozzle divergence angle and the nozzle surface influence separation. Other researchers have determined that the influence of the lag of chemical equilibrium on the nozzle flow is small (1.205). The effect of solid particles in the gas, since they sometimes occur in solid propellant rockets (1.206), and the loss of available energy and pressure in different relative sizes of combustion chambers (1.104) have also been investigated. The effects of altitude on combustion and ignition were found to be less severe in large units (1.207).

1.3 *Flight Analyses*

Although the trajectory varies considerably for different kinds of missiles, the basic principles and methods of analysis are the same (1.301, 1.302, 1.303). Many specialized trajectories, such as simple vertical linear trajectories (1.304, 1.305), motion in the outer atmos-

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[2] Numbers in parentheses refer to References on pages 22–27.

phere (1.306), projectile flight paths (1.307), and antiaircraft missile maneuvers (1.308), have been studied. These analyses have one common characteristic; that is, they permit the evaluation of rocket engine design constants, missile characteristics, or aerodynamic configuration in terms of missile trajectory parameters such as range, maximum altitude, time-to-target, or maximum vehicle velocity. These missile parameters permit a better evaluation of the relative merits of other rocket engine parameters, namely: specific thrust, engine weight, and propellant density for optimum flight performance. A variation calculation of different types of trajectories which will accomplish a given missile mission will permit the determination of an optimum trajectory—one which will give a minimum energy expenditure or a maximum hit probability (1.309, 1.310). Ackeret points out that at extremely high missile speed, one has to include relativistic considerations (1.311).

Because of the large use of unguided rocket projectiles, extensive flight-path investigations (1.312) have studied refinements such as the effectiveness of various fin arrangements (1.313), jet alignment statistics (1.314), and airplane motion during missile launching (1.315).

Although piloted, supersonic rocket flight was realized only in the past few years, many early investigators have analyzed and experimented with its possibilities (1.316). The discussion of supersonic aerodynamics is not within the scope of this paper; however, reference is made to some of the problems relating to missiles. For example, in certain types of missile configurations, the aerodynamic interference effects necessitate unusual control mechanisms for stable flight (1.317, 1.318). Because of the accurate control requirements and the high speeds involved, the guidance and control mechanisms must be very closely correlated with the aerodynamic flight characteristics (1.319, 1.320). Experimental and analytical investigations are also available to determine the aerodynamic pressure at the missile base (1.321). Even at very high altitudes (several thousand miles), it seems that atmospheric forces may determine high-speed flight performance (1.322).

1.4 *Heat Transfer*

While a good deal of progress has been made in the field of heat transfer, the solutions to problems which arise in rocket motors can still be only partly predicted by analysis because the injector pattern and the chamber geometry seriously affect the chamber wall temperatures (1.401, 1.402). Although the larger part of the heat rejection to the wall is by convection, a small part of the heat transfer is by radiation, and its magnitude and effect can be estimated (1.403). The discovery and investigation of the mechanisms of liquid film boiling at high heat-transfer rates not only led to a better understanding of the phenomena in regeneratively cooled rocket motors, but also made possible rocket motor design with high heat-flux densities and found application in other fields (1.404, 1.405, 1.406). Sweat cooling, or the cooling of a porous wall by injection of fluid through the wall, was found to be an effective, but somewhat difficult method of handling propellants of high combustion temperatures at high pressures (1.407, 1.408). Uncooled motor chambers act as a heat sponge and their heat-transfer analysis is a transient problem (1.409).

The aerodynamic heating of the skin of missiles traveling at very high speeds is sufficiently serious to impose severe material problems on the missile, to limit its maximum speed, or to require special skin-cooling devices (1.410, 1.411).

2 Propellants

2.1 *General*

The method of predicting by analysis the performance of any given rocket propellant combination has been perfected in the past ten years (2.101, 2.102). It is based on the knowledge of chemical reaction constants and physical and chemical measurements, many of which were accumulated only recently (2.103, 2.104, 2.105). With these calculations the performance of most common propellant combinations has been accurately computed (1.104, 2.106). This has been an international undertaking with contributions from various parts of the world (2.107, 2.108, 2.109).

Unfortunately, a good fundamental understanding of the complex combustion process is still lacking, but certain types and phases of the combustion process have been investigated empirically (2.110, 2.111). The effect of the lag of chemical reactions in the rocket nozzle on the performance calculations has been found to be small (2.112, 2.113, 2.114). Because of the hazardous, toxic, auto-igniting, or corrosive properties of many of the propellants, their safe handling has presented many practical problems (2.115, 2.116), and safety precautions are now vigorously enforced.

Many propellants have gone from the research phase to the production phase and, judging from the effort expended on the development and production of propellants during World War II, the over-all effort in research and development of propellants during any future emergency will be tremendous (2.117).

2.2 *Liquid Propellants*

The choice of liquid propellants for any given engine is dictated by the application, availability, performance, and properties of the propellant. For this reason, in recent years the effort has been concentrated on learning more of the physical and chemical properties of common and unusual propellants (2.201, 2.202, 2.203, 2.204), on calculating and proof-testing the performance of new possible propellant combinations (2.205, 2.206, 2.207), on learning improved methods of handling and producing propellants (2.208, 2.209, 2.210), on reducing

ignition delays and starting uncertainties (2.211), and on research on materials and rocket engine features which are particularly adaptable to certain propellants (2.212, 2.213). Much work has also been done to improve the logistic and operational characteristics of existing propellants, such as the investigation of additives which lower the freezing point or inhibit deterioration, or the simplification of the chemical production process so as to make the propellants cheaper and more available. Many existing propellant combinations have been thoroughly tested and investigated in order to evaluate more fully their operational uses (2.214, 2.215).

2.3 *Solid Propellants*

The excellent work accomplished during World War II on double-base propellants—burning characteristics, testing, and design problem—has now been documented (2.301). The search for improved propellants with high energy content, good physical properties, good storage characteristics over a wide temperature range, good burning qualities, low temperature sensitivity, and low cost is continuing (2.302, 2.303, 2.304). Possibilities of making the chamber and the charge of plastic material offer low cost fabrication and certain electrical advantages (2.305). Solid propellant charges have been produced in many forms and in sizes ranging from a few ounces thrust (model airplane rocket) to boosters of 50,000-lb thrust (2.306).

3 Research and Instruction

Some of the peacetime applications of rockets are their uses as research tools for aerodynamic measurements at transonic and supersonic speeds, for investigation of the upper atmosphere, and for propulsion of research aircraft (3.001). The British have used a small rocket-propelled model for transonic investigations (3.002), and many of the recent experimental launchings have helped to further research purposes (3.003, 3.004, 3.005).

A large number of technical colleges and research laboratories are actively engaged in theoretical research and instruction in jet propulsion and rocket fundamentals (3.006, 3.007, 3.008). It is encouraging to know that at least six major institutions of higher technical learning in this country are now equipped with rocket test facilities for experiment and instruction. Jet propulsion is now a recognized branch of engineering, and advanced degrees are offered in this specialized field.

The high-altitude research program conducted by the military forces with captured V-2 missiles at White Sands, N. Mex., has furnished a better knowledge of the composition, radiation characteristics, and physical properties of the upper atmosphere (3.009). Such a program created new instrumentation problems, including the measurement of cosmic ray radiations (3.010) and the photography of the solar system from high altitudes (3.011). The literature also shows some of the fruits of research in the upper atmosphere in terms of increased knowledge (3.012, 3.013).

4 Rocket Engines

4.1 *Liquid Propellant Engines*

The general principles, construction methods, and design parameters of liquid propellant rocket engines are well documented (1.104, 4.101, 4.102). The rocket motor (also called thrust chamber), the principal component of the engine, has been designed, built, and tested in many different forms and in thrust sizes ranging from a few pounds to well over 50,000 lb (4.103, 4.104, 4.105). The thrust-to-weight ratio is a good parameter to indicate the progress of rocket motor design; from early values of 3 lb of thrust per lb of weight it has increased to approximately 100 for more recent motors. The mechanism of unstable combustion, which is intimately connected with mixing, heat transfer, and burning of the propellants is not yet fully understood, but attempts have been made to harness these problems into mathematical terms (4.106, 4.107). Some uncooled rocket motors recently have used ceramic linings with excellent motor endurance (4.108, 4.109). The heat transfer analyses discussed above apply to the design of rocket motors because they determine the method of cooling, the coolant velocities, and the design complexity of the units (4.110). The injector design tends to become more empirical, since the detail injector configuration controls the motor performance, heat transfer, and vibration characteristics (4.111).

It is interesting to compare early papers on liquid propellant feed systems (4.112, 4.113) with recent publications on the subject (4.114, 4.115, 4.116, 4.117). While the principal ideas for different types of feed systems were in existence some 20 years ago, the present-day applications of some of these ideas vary greatly in technical conception and detail. Early researchers worked with turbopumps, but they had little to say about bleed-type turbines (bleed gases from the combustion chamber) or about the blast turbines (blades partly immersed in rocket motor flame). The intermittent type of feed system, which was in vogue some 15 years ago, is almost forgotten today. One of the major problems of liquid propellant engines is that of control, which embraces valves and electrical, hydraulic, and pneumatic components (4.118, 4.119, 4.120).

The individual engine design is strongly influenced by the type of propellants to be used, for this governs, among other things, the choice of materials and starting mechanisms (4.121, 4.122, 4.123, 4.124). Thus, a wide variety of liquid propellants has been investigated experimentally on the test stand, and many have seen actual service.

With the progress of knowledge in this field of liquid propellant rocket engines, attention seems to focus

more and more on practical usage problems, such as reliability features, simplicity of control and construction, servicing difficulties, etc. Many, but not all of the fundamental problems are solved, and research and development are now more concentrated on modifications, improvements, and adaptation to different propellants or different design conditions.

4.2 *Solid Propellant Engines*

The solid propellant rocket is the oldest form of rocket and still the most widely used as assisted take-off units, boosters, and projectile power plant (4.201, 4.202). The problems of long storage life with chemical or physical deterioration, wide temperature operating limits, and stable burning are being solved by persistent research (4.203) and development testing (4.204). Because these rocket motors are now produced in relatively large quantities and because they are used as projectiles where accuracy of weight and alignment is important, the metal parts fabrication method has to be carefully selected and controlled (4.205, 4.206). Thus, the emphasis has shifted with the progress from its early research phases to problems of mass production and operational use.

4.3 *Atomic Rocket Engine*

Since the performance of rocket engines with chemically reacting propellants is limited to a specific thrust value of somewhat under 400 lb-sec per lb, investigators have looked for other sources of energy. Atomic power provides the possibility of a two- to threefold increase in performance if propellants of low molecular weight are used (4.301). Several modifications of an atomic rocket have been analyzed: Some in which a working fluid such as hydrogen or water is heated; some in which neutrons are ejected (4.302, 4.303). Shielding provisions and fuel problems receive special attention (4.304, 4.305), and in some preliminary design it appears that the shielding requirement imposes too much weight on large bombers (4.306). Thus, the atomic rocket power plant is still in its infancy, and the literature does not yet show any evidence of experimental engine test results.

5 Guided Missiles and Rocket Projectiles

The guided missile is relatively young. The term came into general use in open publications only in 1947. This new weapon, which comes in many forms (air-to-air type, ground-to-air, air-to-ground, and ground-to-ground) uses rocket engines in many instances. It is, in fact, one of the principal present-day applications of rocket engines, both large and small, using both liquid and solid propellants (5.101, 5.102). To make it an integral effective weapon, the guided missile has to combine, in an optimum fashion, principles from the science of aerodynamics, mechanics, structures, electronics, optics, thermodynamics, metallurgy, plastics, and many others. The literature is varied and ranges from discussions of trajectory dynamics (5.103, 5.104) and flight stability (5.105, 5.106, 5.107), to problems of guidance and control (5.108–5.111), homing devices (5.112), and recovery methods (5.113). These typical examples of the literature bear out the complexity of the devices. This complexity, which is necessary to achieve the required objectives of performance, accuracy, and maneuverability, is also one of the biggest obstacles from the standpoint of reliability and ease of fabrication and development.

The Germans were the first to use guided missiles effectively and they had a variety of different types (5.114, 5.115, 5.116). Recent publications list some British (5.117) and American types (5.118, 5.119). Literature references to production (5.120) testify to the completion of the development phase of some of these missiles.

Unguided missiles and projectiles do not have the complexity and glamour of the guided type, and the literature is not as voluminous on this subject. A variety of different types has been developed, ranging from the bazooka to some large types (5.201, 5.202). Their basic problems of propulsion and flight are being vigorously attacked and solved (5.203, 5.204).

Accuracy of thrust alignment and fabrication are particularly important in such applications (5.205). The larger part of a 482-page book is devoted to the history of problems of wartime solid propellant projectile developments (5.206).

6 Rocket Aircraft and Assisted Take-off Units

The Germans also pioneered in the development of rocket-powered, piloted aircraft with their ME 163 and other fighter planes (6.101, 6.102). Two U. S. planes (the XS-1 and the D558-2) are known to have been rocket-driven (6.103, 6.104). The principal problems of developing aircraft rocket power plants are in obtaining absolute reliability and stable operation with throttled, variable thrust, and satisfactory starting at high altitude. Thus, this type of rocket engine has a relatively complicated starting and control system (6.105, 6.106, 6.107).

A rocket-powered aircraft has the advantages of excellent ceiling, speed, and climb characteristics if compared to airplanes with other power plants (6.108, 6.109); however, its high specific fuel consumption limits its range, its peak performance, and its flight duration to a few minutes (6.110, 6.111). Means for increasing the range do not appear very promising (6.112). While piloted rocket-powered aircraft have some performance advantages over other means of aircraft propulsion (6.113), it appears that the actual mission of split-second enemy interception can be accomplished more effectively by a rocket-powered guided missile. In this case, the pilot's primary function would be to return the aircraft or missile to its base. Thus, it appears that the rocket interceptor is a close relative of the anti-aircraft missile.

The power boost given to conventional aircraft by assisted take-off units permits take-off with a heavier load or in a shorter landing-strip distance (6.201, 6.202, 6.203). Because of the low propulsive efficiency at conventional aircraft speeds, no particularly large increase in aircraft performance is feasible when using these units in flight. However, at high altitudes, conventional turbojets, turboprops, and piston engines lose power, and there an auxiliary rocket engine shows considerable advantage. Typical examples are the liquid-propelled British Sprite (6.204) and a BMW design of a rocket booster, which draws pumping power from the main jet engine (6.205). These two units are complicated, relatively expensive, but suitable for repeated use. A simpler, solid propellant droppable unit has been used in commercial aircraft applications (4.201).

7 Military Aspects

Almost all of the rocket work done today is for military purposes. The commercial applications are apparently small (7.001). Thus, the cognizant military agencies have four basic tasks:

1 To prove and evaluate the tactical and operational effectiveness of present and future rocket weapons. This is accomplished by elaborate tests under severe operational conditions (7.002, 7.003).

2 To decide the course and direction of future research, development, production, and use of rocket weapons; and to decide the emphasis which should be placed on one type compared with another (7.004–7.007).

3 To support and stimulate an industry capable of developing good guided missiles and reliable rocket engines and producing them in case of emergency (7.008). In this respect, the U. S. military services have done a creditable job. One reference estimates that in 1950 there were more than 4000 skilled people and more than 12 companies engaged in the rocket engine field. Considering that ten years ago there was no such industrial activity, this speaks well for the advance of rocket propulsion.

4 To create and train military units which will operate rocket engines or use rocket weapons (7.009, 7.010). This is, in itself, a formidable task. It took the Germans approximately one year—and many expended missiles—to train some 3000 soldiers in the handling and launching of the V-2.

Not all the rockets used for military purposes are found in missiles and aircraft; there are also rocket-driven torpedoes (7.011) and turbine starters (7.012).

8 Space Travel

Ever since the early researchers realized that the rocket engine offers the fascinating possibility of escaping from the earth, men have calculated, speculated, and investigated means for actually accomplishing this feat. The literature which has been published in the past 30 years on space travel is unique in two respects: In the first place, it is relatively voluminous compared to the literature covering other phases of rocket propulsion. The majority of bound books deal fully, or to a large part with escape from the earth or interplanetary travel, many in a semitechnical style. Several of the classical works on rocket propulsion fall into this group. In the second place, the authors of these publications on space travel come from many different countries, and, contrary to many other phases of the rocket literature, very few authors are from the United States. One British expert attributes this condition in the United States to its more active and practical experimental and research efforts, which promote more literary work on current investigations. Some typical publications are listed under references 8.001–8.010.

While a good part of the early works on space travel were nontechnical and even fictional, it is encouraging to find later art on a higher technical level with typical detailed mathematical solutions of space-travel trajectories (8.011–8.015) and with preliminary design investigations of multistep rocket space ships (8.016, 8.017). Some investigators have made cost estimates of such an undertaking (8.018, 8.019). One reference states that the cost of a space station of well over 250 million dollars can easily be amortized by the income from communication and television rights, weather observation, and advertising revenue. It is also encouraging that recent investigations have dealt with practical problems of space travel, such as aeromedical problems and meteors (8.020, 8.021, 8.022). It seems that we are getting closer to the real problems of space travel which, various authors predict, may take place as early as 1960, or not later than the year 2000.

9 Tests and Operations

9.1 *Facilities*

As in other phases of rocketry, new techniques and procedures for testing and operating had to be developed. Since the experimental development of rockets is accompanied by occasional fires and explosions, emphasis has to be placed on personnel safety in the construction of test stands (9.101, 1.104, 9.102). At the same time, the test facility design has to be such that its working efficiency and versatility are not seriously affected by these safety features (9.103). Flight tests must necessarily be conducted at isolated locations so that runaway missiles will not cause harm. The construction of adequate test facilities, housing, servicing equipment, and flight-path tracking devices in isolated areas presents considerable difficulties (9.104, 9.105, 9.106).

The launching of each rocket missile and the flight preparation of rocket aircraft is a major effort involving a considerable number of skilled personnel and special

fueling and calibrating equipment (9.107, 9.108, 9.109). Since guided missiles are often designed for flight loads only, special ground-handling equipment (9.110) is required, such as, for example, the carriage used in transporting and erecting the V-2.

9.2 *Instrumentation*

The testing of rockets required the development of special instruments capable of measuring thrust, pressures, high liquid flows, temperatures, and other quantities. A high frequency response has been found to be essential to measure variable phenomena which occur only for a very short time, and the automatic recording of data has been found to be much more satisfactory than observation of indicated gages. Thus, a variety of new instruments had to be evolved, many of which have found application in other fields of endeavor (9.201–9.205).

Furthermore, flight testing posed particular problems in recording data through the medium of telemetering (9.206), or in recording data inside special shockproof armored cans designed to survive the missile impact (9.207). The tracking of the actual flight path in experimental missiles is accomplished by radar and/or optical means and permits determinations of variations in accelerations, velocities, and distances in all three-dimensional coordinates (9.208–9.211).

While the extra complications of test-stand safety features, engine safety devices, and special instrumentation are definite requirements during the research and development test phases, they are hindrances to production-testing and field-launching. Further, by looking at the preparation necessary to fly the average rocket-powered airplane or missile, one finds today's launching procedures cumbersome, and it seems that a good deal of work needs to be done to simplify them. The relatively slow progress made in overcoming these difficulties is due in part to the lack of having extensive field experience with large numbers of complicated rocket engines, such as the Germans were able to accumulate with their V-2.

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The 1951 ARS Convention: A Technical Survey

(*Continued from page 6*)

Variations in jet length, diameter, impingement angle and velocity were attempted, and a photographic record was obtained for analysis. It was found that upon impingement, the two jets formed a ruffled sheet of liquid which periodically disintegrated, forming irregularly spaced waves of varying intensity. The frequency of wave formation was constant over a finite time interval under constant operating conditions. The most important factor affecting spray frequency was found to be the sheet velocity after impingement.

In the discussion period following the paper, K. D. Miller of the M. W. Kellogg Company, stated that wave fluctuations were also observed in their splash plate studies. He expressed the opinion that there is a possible correspondence of cold jet and combustion instabilities that could, in part, control fundamental rocket instability frequencies.

It was most unfortunate that many members and guests could not remain for the last two sessions to actively participate in the interesting and controversial presentations.

EDITOR'S NOTE: A more general account of the activities of the Annual Convention, including reports on the Business Meeting and the Honors Night banquet, appears in the American Rocket Society News section in this issue, page 43.

Rocket Applications of the Cavitating Venturi

By L. N. RANDALL[1]

Curtiss-Wright Corporation, Caldwell, N. J.

The principles of operation and details of construction of this very interesting device are presented. Its development and application to liquid propellant rockets are described. It has been employed extensively as an extremely simple and accurate flow control. Its usefulness as a temperature device is also outlined, as are special forms designed to meet particular requirements.

FOR THE past few years the Curtiss-Wright Corporation has used the cavitating Venturi in its rocket test program as an extremely simple and accurate means for controlling flow of incompressible fluids.

Although the interesting phenomenon that makes this possible is apparently not new, it has to the best of the writer's knowledge been employed only in rocket applications. Its principle of operation may be expressed thus: As the pressure drop across a conventional Venturi is increased, a point is reached at the throat where substantially all of the upstream head is converted into velocity head. The only static head remaining is that of the fluid vapor pressure. If, under these conditions the upstream head is maintained constant, a further increase of the pressure drop obtained by decreasing the downstream pressure cannot result in increased flow. This characteristic of a Venturi has been used to advantage in liquid propellant rockets and may find applications in other than the rocket field.

Unusual flow characteristics of thick-plate orifices exhibited early in this company's test program led to the evolution of the "cavitating Venturi" concept and its application to the control of propellant flows. It may be of interest to examine briefly this early experience with thick orifices having sharp entrances. These orifices were chosen because they provided a simple method of adjusting pressure drop within a rocket hydraulic system, and they could be sized at the test site from the information gained from water calibrations. Because these water calibrations were made of the complete rocket motor hydraulic system as a unit, and because of facility limitations, it was necessary to allow the system to discharge to atmospheric pressure. Although this is generally considered bad practice when calibrating, a few check runs indicated this method to be sufficiently accurate and gave no trouble until later in the test program. In time discrepancies began to appear in the "hot" testing data when compared with those of the water calibration. Assuming all phenomena were being considered, this indicated rather inaccurate data, which was not considered a satisfactory answer. Further investigation of the problem indicated the combination of the orifices used and the calibrating back pressures to be at fault. A typical orifice used at this time is illustrated in Fig. 1. The curve there shows a typical pressure/flow relationship when an orifice of this type is calibrated at low pressure. It will be noted that appreciable change in pressure drop across the orifice in one region produced no change in flow. This lack of change in flow caused the wide discrepancy in the data. While the immediate remedy in the test program was to calibrate with high back pressure on the rocket motor, this phenomenon created considerable interest. The only apparent solution was that the flow was limited by the size of the vena contracta where the total upstream head had been converted to velocity head, a portion of which was recoverable in the downstream part of the orifice. However, the diffuser efficiency of this straight section of the orifice being low, constant flow could be maintained over only a relatively small range of downstream pres-

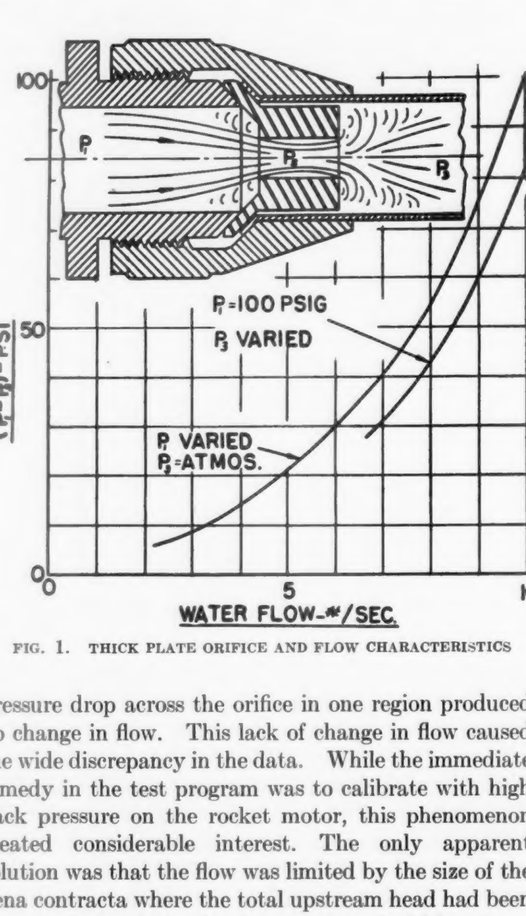

FIG. 1. THICK PLATE ORIFICE AND FLOW CHARACTERISTICS

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[1] Rocket Test Supervisor. Member ARS.

sures. The obvious step to improve this was to construct an efficient diffuser section, at which point the device took the form of a conventional Venturi. It was thus found possible to extend the cavitating region up to a point where the downstream pressure approached 80–90 per cent of the absolute upstream pressure. Expressed in equation form, and ignoring that small amount of fluid that will be found in vapor form while cavitating, the familiar Bernoulli equation applies, and the flow relationships are as follows:

$$h_1 + \frac{V_1^2}{2g} = h_2 + \frac{V_2^2}{2g} + \text{Losses (1–2)} = h_3\,\frac{V_3^2}{2g} + \text{Losses (2–3)}$$

where

h = pressure head in ft
V = velocity head in ft/sec
g = gravitational constant

If 1 refers to the throat section upstream of the Venturi and 2 to the throat, it is found that h_2 reduces to the vapor pressure of the liquid. The pressure will be maintained at this level by the liquid vapor pressure because of the change of state of the fluid at its interface with the Venturi wall. This phenomenon, called cavitation, is found to exist whenever the total head at the end of the Venturi diffuser 3 is less than approximately 85 per cent of the upstream head and more than the vapor pressure of the liquid.

This leads to the first and probably most important application of the cavitating Venturi—i.e., it is an extremely simple flow control. Since V_2 and the area at the throat remain constant, the flow remains constant depending only on upstream head and the liquid vapor pressure where the recovered total head does not exceed the recovery factor of the diffuser. In practice, upstream head and vapor pressure usually vary over only an insignificant range, while downstream pressures need not be accurately determined to predict the exact flow.

Fig. 2 pictures a transparent Venturi flowing water and operating as a flow control (a) with the back pressure approximately 85 per cent of upstream pressure, and (b) with the back pressure approximately 10 per cent upstream pressure. In both cases the water flow is the same. However, it will be noted that in (a) a very short region of cavitation exists at the throat while in (b) the cavitating region has been greatly extended with only a short portion of the diffuser being effectively used. In the case of (b) that portion of the original head not required is lost in turbulence and appears as heat and will not be recovered. Although not visible in Fig. 2, the vapor is actually a shroud surrounding a solid jet of liquid moving at high velocity as illustrated in Fig. 3.

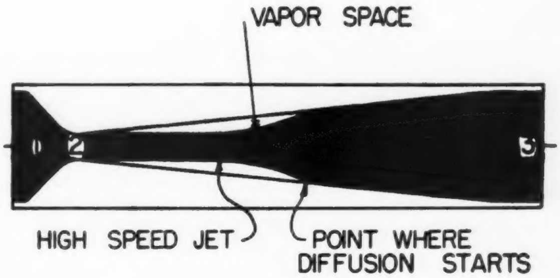

FIG. 3 VENTURI OPERATION WHILE CAVITATING

In the design of such a Venturi to be used as a flow control, the designer takes into consideration the total head available and the vapor pressure of the liquid. When these have been established, the difference between these two will be the total static head available for conversion into velocity. In other words, this is the differential pressure or head used to find the velocity using the familiar equation

$$h = \frac{V^2}{2g} \text{ or } V = \sqrt{2gh}$$

After obtaining the velocity and assuming an orifice coefficient (conventional design usually gives a C_D of from 0.96 to 0.98), the throat area is obtained.

$$A = \frac{\text{Flow Rate}}{VC_D}$$

With the throat established the designer then uses conventional diverging and converging sections to complete the Venturi with a nicely rounded transition at the throat. Where space is limited it is better to make the approach more abrupt and retain a maximum diffuse divergence angle of 5–6 deg. If the diffuser must be shortened, it is better to cut it off at the downstream end, rather than to use a wide angle cone.

Generally speaking, a well-designed Venturi used in a cavitating manner as a flow control will cavitate and maintain a throat pressure equal to the vapor pressure of the liquid at its initial temperature as long as both the inlet pressure and the outlet pressure of the Venturi are above the vapor pressure of the liquid, and the outlet pressure does not exceed approximately 85 per cent of the initial pressure.

This is not to be confused with the cavitating char-

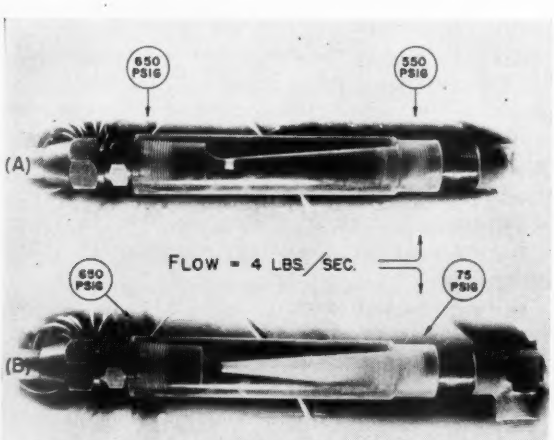

FIG. 2. TRANSPARENT VENTURI FLOWING WATER AND OPERATING AS FLOW CONTROL

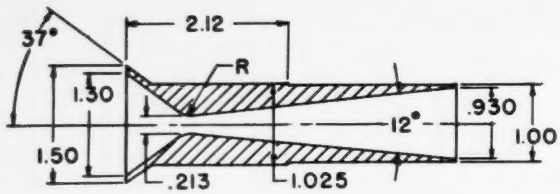

FIG. 4 TYPICAL CAVITATING VENTURI ORIFICE

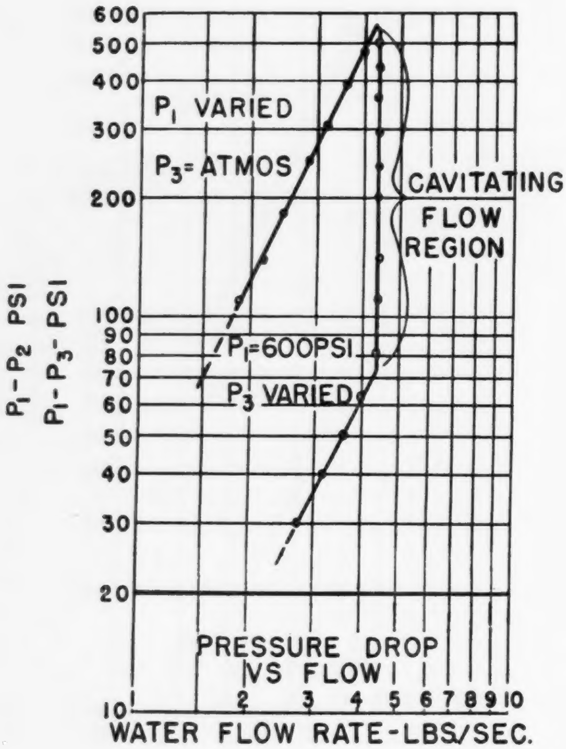

FIG. 5 TYPICAL CAVITATING VENTURI CALIBRATION

acteristics of nozzles or Venturis discharging into an atmosphere equal to or less than the vapor pressure of the liquid at its initial temperature.

The term 85 per cent of upstream pressure is used in a general sense, as the exact point at which cavitation will occur depends upon efficiency of the diffuser section of the Venturi and other factors such as density, viscosity, the cohesion and adhesion properties of the liquid, and dissolved gases within the liquid. The diffuser efficiency for various flow relationships of the Venturi must be determined from actual calibration.

Exploitation of this flow-controlling characteristic of cavitating Venturis at Curtiss-Wright has been for the most part limited to its use for test purposes or to engine applications, where a 15 per cent loss in head required for control purposes can be tolerated without imposing excessive demands on pressurization sources. That this use has been extensive can be appreciated from the fact that some 200 of these Venturi orifices have been calibrated and are in use. A typical construction is illustrated in Fig. 4. Units employed are designed to fit directly in standard tubing connections of various sizes. Each is sized by calculations and then calibrated with water against accurate laboratory flow instrumentation, with the calibration being made a part of the permanent record for use by engineering, test, and data reduction personnel. Typical calibration data are plotted in Fig. 5. The lower line is a calibration of a given Venturi with no cavitation at the throat showing $P_1 - P_3$ vs flow. The upper line represents the low limit of cavitation and is a plot of $P_1 - P_2$ vs flow. P_2 equals the vapor pressure of water at 70 deg, but for practical purposes is assumed to be zero when working with P_1 values of over 100 psia. A point (when plotting $P_1 - P_3$ flow) falling between these two lines will result in cavitation and the resulting flow-control characteristics described. For example, with a P_1 of 600 psia and a P_3 value of 100 psia, we find we are in the cavitating region. At the same value of P_1, we find we can increase P_3 without changing flow until we have reached a value of 520 psia. With further increases of P_3, the flow is found to drop along the low line indicating cavitation no longer exists.

These Venturi orifices have been found to be particularly helpful in the development of thrust chambers, injectors, gas generators, etc. With them it is possible to predetermine accurately the amount of fluid introduced into the system or part of a system without complicated flow-control equipment and without detail knowledge of back pressures built up by such relatively unpredictable phenomena as, e.g., combustion efficiency.

It will be seen that the cavitating Venturi acting as a flow control has the following limitations: (*a*) The maximum back pressure cannot exceed 85 per cent of the upstream head—this means the loss of better than 15 per cent head must be tolerated for flow-control purposes. (*b*) Without the addition of moving parts to vary the throat area, the flow can be changed over only a relatively limited range in normal applications, because the flow varies as the square root of the upstream head and any change in flow requires a substantial change in upstream pressure. (*c*) Under varying downstream pressures there is a transient variation in flow of extremely short duration resulting from the change in vapor volume when the diffuser adjusts automatically to its proper value of recovery.

In order to increase the flexibility of application, some of the larger sizes of Venturis have been constructed with adjustable pintles to vary the effective throat area. A typical one is illustrated in Fig. 6. The use of a calibrated pintle allows the flow to be reduced to as low as 30 per cent of the original value with only a slight reduction in accuracy and some drop in diffuser efficiency. Maximum diffuser recovery may

drop as low as 75 per cent of the upstream head when the pintle offers its greatest restriction. Surprisingly enough, the cavitation has not produced any noticeable erosion on the throat or pintle in these Venturi orifices. However, trouble has been experienced when an effort was made to control the pintle from the downstream end of the diffuser tube. The high turbulence in the cavitating region is likely to produce destructive vibrations in the pintle and supporting elements. Fig. 7 illustrates a typical calibration of a variable Venturi.

A word of caution to the experimenter: When calibrating a cavitating Venturi using a liquid pressure system having high pressure gas as the source of pressure, a fair amount of the gas is dissolved in the liquid as it stands. This gas is released in the throat of the Venturi and will cause error if the flow measurement is taken at a point downstream of the Venturi. Where possible, the calibrating instrument should be placed upstream. When this is not practical the calibrating instrument should be operated under as much suppression head as possible to allow the gas to again dissolve before attempting to meter the liquid flow. In this manner, errors arising from the release of the dissolved gases are minimized.

An extension in the application of cavitating Venturis arises in the metering of fluids which under test conditions undergo a considerable change in temperature with attendant major changes in density. An example is liquid oxygen. By introducing a pressure tap slightly downstream of the throat of a cavitating Venturi, the liquid vapor pressure existing at the time of the test can be measured. From these data the temperature and density variation can be calculated and used in correcting the flow values. This has proved particularly useful in providing accurate temperature data for transient conditions of short duration.

Still a third use for the cavitating Venturi is as a means of detecting the difference between the presence of a liquid or a gas flowing in a line at equal upstream pressures. This application has proved useful, when used in pressurized liquid systems, for terminating a test run when all of the liquid in the tank has been used, but before the pressurizing gas has entered the combustion chamber. The device consisted of actuating a pressure switch connected to a throat tap of the Venturi by the increase in pressure attending the change from liquid to gaseous flow. Operation of the pressure switch triggered the shutoff of the test equipment. Since the cavitation pressure is in the neighborhood of one atmosphere, and the pressure with gaseous flow usually about half the upstream pressure (critical flow for a compressible fluid), there is ample pressure differential to insure consistent operation of the pressure switch.

The foregoing touches only on the more important applications of the cavitating Venturi as used in the past. The possibilities for applications in the rocket and other fields are limited only by the imagination of the reader.

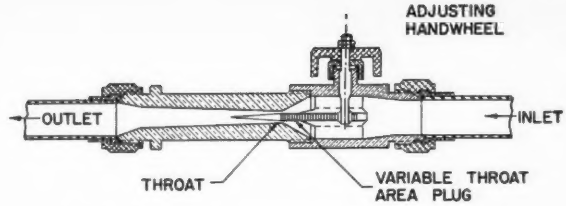

FIG. 6 VENTURI—VARIABLE ORIFICE FLOW CONTROL

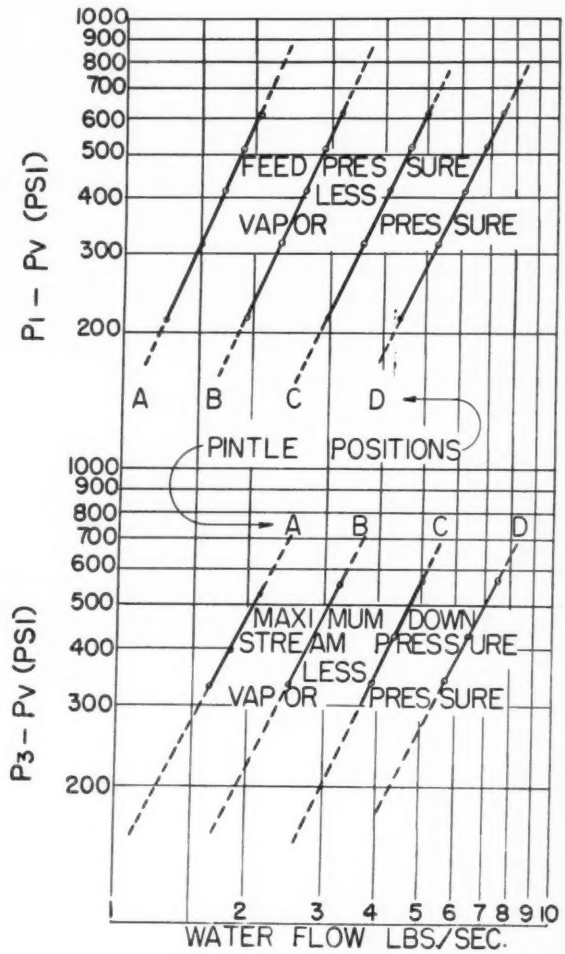

FIG. 7 TYPICAL VARIABLE VENTURI ORIFICE CALIBRATION

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(*Continued from page 27*)

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The Effects of Several Variables Upon the Ignition Lag of Hypergolic Fuels Oxidized by Nitric Acid[1]

By STANLEY V. GUNN[2]

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A method is described for measuring the ignition lag of self-igniting (hypergolic) bipropellant combinations. Ignition lag data are reported for combinations of nitric acid with aniline, furfuryl alcohol, and mixtures of aniline and furfuryl alcohol. The ignition lags ranged from about 10 to about 400 milliseconds, depending upon such variables as temperature, acid composition, fuel composition, and metallic additives.

Introduction

THE determination of the ignition lags of various hypergolic liquid rocket propellants has been the object of considerable research effort by a number of rocket research agencies. Several ingenious devices (1, 2)[3] have been developed for measuring the ignition delay period, but most of them are described in classified documents. One of the tests currently employed in the aforementioned determinations is the open cup test. Because the time intervals to be measured are generally quite short, special techniques based on either electronic devices or high-speed photography are employed. The accuracy of the measurement of the ignition delay period depends upon the ability of the time-measuring system to sense the phenomena selected for defining the beginning and end of the time interval constituting the ignition lag. This paper presents some of the results obtained with one type of open-cup testing apparatus.

Description of Apparatus and Test Procedure

An electronic timer has been developed at the Purdue Rocket Laboratory (3), and the timer in conjunction with certain associated reaction apparatus has been employed for ignition lag measurements in open-cup tests. Fig. 1 is a drawing of a partial assembly of the essential elements of the chemical reaction apparatus employed in the open-cut tests. The elements are (1) a constant temperature, molded glass, reaction dish; (2) a weir-lipped cup; and (3) a support stand equipped with a mechanism for pouring one propellant into the reaction dish containing the other propellant. The reaction apparatus is employed in the following manner: A measured amount of the liquid oxidizer is placed in the cavity of the molded glass reaction dish, and a measured quantity of fuel is placed in the weir-lipped cup; the latter is rotatably mounted, on the support stand, above the reaction dish. When the weir-lipped cup is rotated, the fuel pours out of it in the form of a thin sheet into the reaction dish, mixes with the oxidizer, and chemical reaction is initiated.

The ignition lag is defined as the time interval elapsed between the instant that the fuel stream strikes the surface of the pool of oxidizer and the instant that visible light is emitted from the reacting propellants.

Fig. 2 illustrates schematically the electronic timer employed for measuring the ignition lag; the timer comprises the following: (*a*) pulse generator—cathode follower, (*b*) single sweep generator, (*c*) cathode-ray oscillograph, (*d*) signal generator, (*e*) phototube and phototube amplifier, and (*f*) oscillograph-record camera.

Fig. 3 is a circuit diagram of the pulse generator-cathode follower unit. It is a single-stage amplifier with the control grid terminal, middle A, connected to an electrode immersed in the pool of oxidizer and the plate terminal, top A, connected to the weir-lipped cup containing the fuel. Any one or a combination of the following three effects produces a positive voltage pulse on

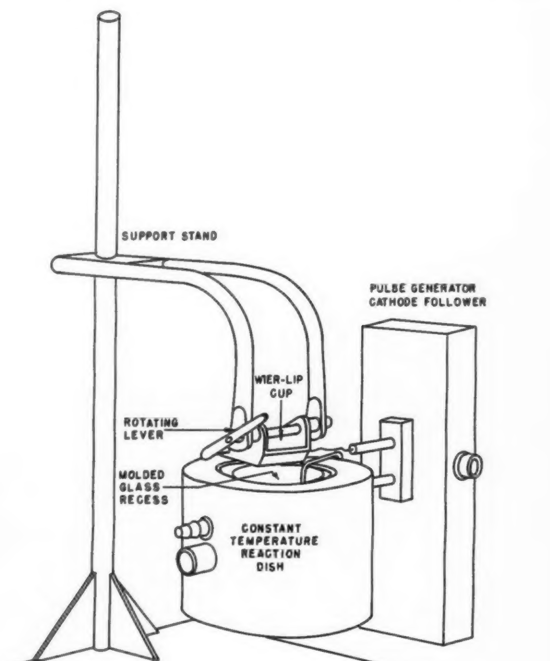

FIG. 1 PARTIAL ASSEMBLY OF CHEMICAL REACTION APPARATUS

[1] Presented at the Fall Technical Session of the AMERICAN ROCKET SOCIETY on Sept. 26–28, 1951.

[2] Research Assistant, Mechanical Engineering Department.

[3] Numbers in parentheses refer to References on page 38.

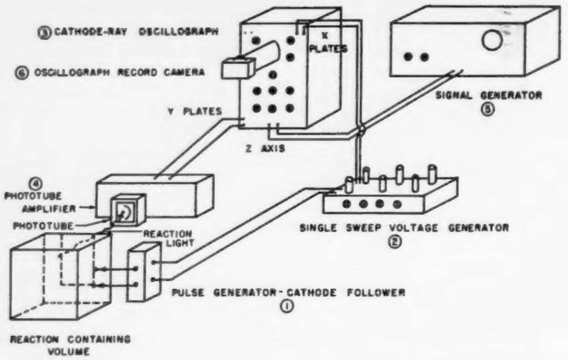

FIG. 2 SCHEMATIC DIAGRAM OF COMPONENTS OF ELECTRONIC IGNITION LAG TIMER

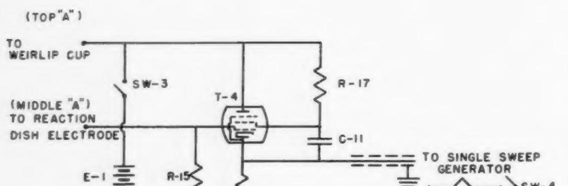

FIG. 3 PULSE GENERATOR-CATHODE FOLLOWER CIRCUIT

the control grid electrode of the pentode vacuum tube T-4, coincident with the instant that the fuel stream leaving the weir-lipped metal cup strikes the surface of the pool of oxidizer in the glass reaction dish. If the fuel is somewhat conductive, the first effect is to cause the control grid suddenly to become more positive, due to the more or less instantaneous decrease in grid-to-plate resistance. Next, because the fuel is actually charged to plate voltage, there will be a mass carry-over of positive charge by the fuel down onto the control grid electrode. The physical arrangement of the electrically charged metal cup suspended over the pool of oxidizer constitutes a capacitive system. Consequently, when the fuel pours out of the metal cup onto the surface of the pool of oxidizer, the capacity of the system is increased causing the control grid to become more positive. The net result of the aforementioned effects is to produce a small triggering voltage pulse across the cathode-biasing resistor, R-16.

The triggering voltage pulse is conducted through a coaxial cable to the single sweep generator. On receipt of the triggering pulse, the single sweep generator causes the electron beam in the cathode-ray tube to be deflected horizontally at a uniform rate, so that a horizontal trace is swept out across the screen of the tube. The length of trace from its initial position to any arbitrary point is a measure of the time elapsed between the generation of the triggering pulse and the instant when the trace reaches the arbitrary point.

A phototube and phototube amplifier sense the emission of the first light quanta from the reacting propellants and generate a sharp voltage pulse coincident with that emission of light. The phototube is resistance-capacitance coupled to the input terminals of the phototube amplifier; the latter is an R/C coupled, wide band, a-c amplifier, possessing two shaping stages of differentiation and amplification, and is terminated with a cathode-follower stage. Whenever there is a change in the intensity of the light received by the phototube, a sharp voltage pulse is produced across the output terminals of the amplifier. The voltage pulse, resulting from the initiation of flame in the reacting propellants, is conducted to the Y-axis terminals of the cathode-ray oscillograph through a coaxial cable. After further amplification within the oscilloscope, the voltage pulse is impressed upon the Y-axis deflection plates of the cathode-ray tube and causes a vertical deflection of the electron beam. The length of trace between initial position and point where the first vertical deflection occurs defines the ignition lag for the propellants.

The length of trace is resolved into units of time by Z-axis modulation of the electron beam from the sine wave output of a signal generator. The signal generator cuts the electron beam "off" and "on" in accordance with the preset frequency of its output signal and causes the timing tracer to appear as a dotted line; the distance between successive dots equals the period of the modulating signal. The image of the timing trace is recorded photographically by means of an oscillograph record camera.

Fig. 4 is an oscillogram illustrating a typical ignition lag determination; the propellants tested were white fuming nitric acid and a fuel mixture of 80 per cent furfuryl alcohol, 20 per cent aniline (by weight). The Z-axis modulation frequency was 2000 cps so that the distance between successive dots is equivalent to 0.5 millisec. Since there are 29 dots on this oscillogram, the ignition lag for the aforementioned propellants amounts to 14.5 millisec at room temperature.

The ignition lag has been previously defined as the time interval elapsed between the instant that the fuel

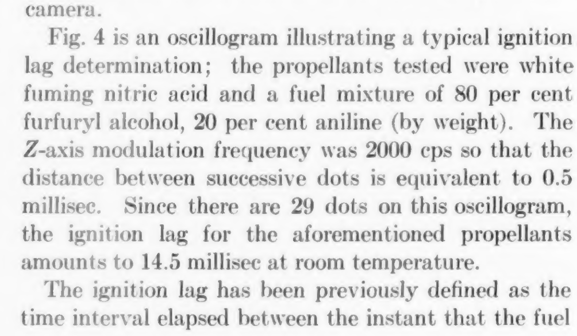

FIG. 4 REPRESENTATIVE OSCILLOGRAM OBTAINED IN DETERMINING THE IGNITION LAG OF HYPERBOLIC ROCKET PROPELLANTS

stream strikes the surface of the pool of oxidizer and the instant that visible light is first emitted from the reacting propellants. By proper selection of the total sweep period of the timing trace and of the frequency of Z-axis modulation, the accuracy of the ignition delay determination can be held to within 2 per cent of the true value of the ignition lag as defined above.

To study the effect of the temperature of the propellants prior to mixing on their ignition lag, hereafter called the mixing temperature, the glass reaction dish was designed so that a coolant can be circulated around the oxidizer pool contained within it to maintain the oxidizer at the desired test temperature. The construction of the weir-lipped cup does not lend itself to cooling the fuel in the aforementioned manner. The fuel temperature is brought to the desired test temperature by first prechilling the weir-lipped cup containing the fuel to a temperature lower than the test temperature; the propellants are reacted when the fuel has warmed up to the temperature of the oxidizer. Iron-Constantan thermocouples are employed for measuring the temperatures of the liquids.

The Effects of Temperature, Additives, and Chemical Purity on Ignition Lag of Liquid Rocket Propellants

Fig. 5 presents data showing the effect of the temperature of the propellants upon the ignition lag of the RFNA-aniline bipropellant system (3, 4). Over a rather wide range of moderate temperatures, the temperature of the propellants appears to exercise little effect upon the ignition lag; but as the test temperature approaches the freezing temperature of the aniline, a marked increase in ignition lag occurs. It appears that a factor contributing to the increase in ignition lag, in addition to the effect of temperature upon the kinetics of the chemical reaction, is the marked increase in the viscosity of the aniline which results in less favorable mixing.

Fig. 6 presents the effect of temperature on the ignition lags of furfuryl alcohol and of a fuel mixture of c.p. 80 per cent furfuryl alcohol, 20 per cent aniline (by weight) oxidized with WFNA (5, 6). It is apparent that the fuel mixture of 80/20 FA-AN possesses superior ignition characteristics to that of the furfuryl alcohol when these fuels are oxidized with WFNA. Other significant features exhibited by the curves are the asymptotic increases in ignition lag as the mixing temperatures approach the freezing temperatures of the respective fuels. The addition of aniline to furfuryl alcohol prevents the fuel mixture from becoming overly viscous until lower mixing temperatures than those possible for the furfuryl alcohol are reached, and the improved mixing resulting from this effect appears to be partially responsible for the delay of the asymptotic increase in ignition lag to lower temperatures.

The presence of impurities or additives in either the oxidizer or the fuel may produce a marked effect upon

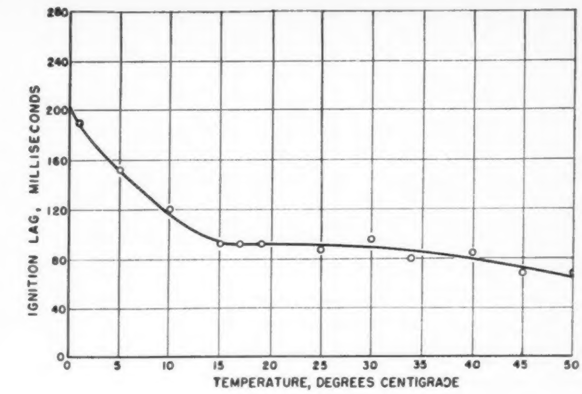

FIG. 5 COMPONENTS OF ELECTRONIC IGNITION LAG TIMER

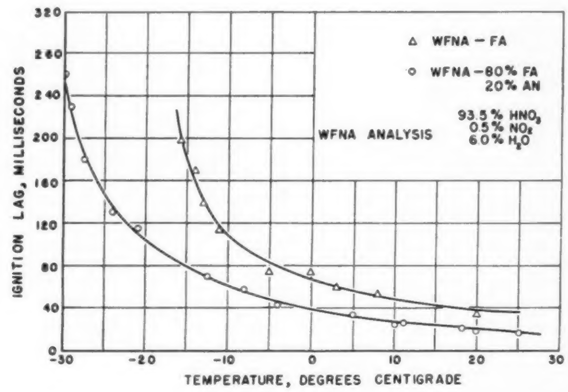

FIG. 6 THE EFFECT OF MIXING TEMPERATURE UPON THE IGNITION LAG OF THE ROCKET PROPELLANTS WFNA-FA AND WFNA-80% FA, 20% AN

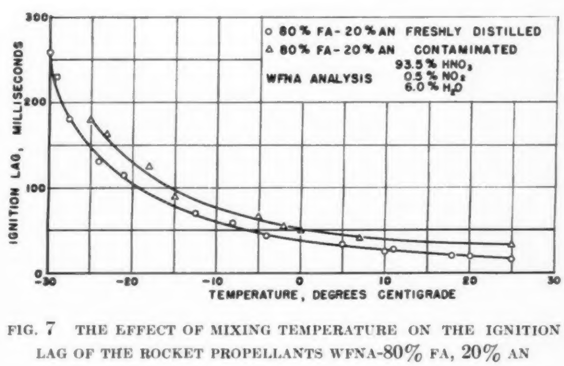

FIG. 7 THE EFFECT OF MIXING TEMPERATURE ON THE IGNITION LAG OF THE ROCKET PROPELLANTS WFNA-80% FA, 20% AN

the ignition properties of the bipropellant system concerned. Fig. 7 presents the effect of propellant temperature upon the ignition lag of 80/20 FA-AN, compounded from furfuryl alcohol and aniline contaminated with gasoline, oxidized with WFNA (7); for comparison, the curve for chemically pure 80/20 FA-AN, presented in Fig. 6, is included in the figure. The effect of gasoline contamination of the fuel mixture was to shift

the ignition lag-temperature curve upwards; i.e., the ignition lag is increased at all propellant temperatures.

The effects of adding selected metals, metallic salts, and a metallic oxide to the oxidizer upon the ignition lag of a fuel mixture of chemically pure 80/20 FA-AN oxidized with WFNA have been studied (7, 8). The additives were dissolved in the acid in the amount necessary to produce 0.5 *N* solutions in WFNA based on the valence of the particular metal in the additive concerned; the analysis of the WFNA used was 93.5 per cent HNO_3, 0.5 per cent NO_2, and 6 per cent H_2O. The metallic salt additives contained a certain amount of water of crystallization, and their addition to the acid resulted in a slight increase in the water content of the acid. All of the additives tested with the aforementioned bipropellant system produced longer ignition delay periods than those obtained for the same propellants without additives. Table 1 presents the measured ignition lags obtained with and without additives when the temperature of the propellants prior to mixing was approximately 70 F.

TABLE 1

Additive	Ignition lag (millisec)	Additive	Ignition lag (millisec)
No additive	18.	$NiCl_2$	25.
V_2O_5	20.5	$CoCl_2$	30.
Cu	22.	KBr	31.
$CuCl_2$	23.	$Cr(H_2O)_6Cl_3$	34.
Fe	23.5		

The addition of pure iron to WFNA produced a slight increase in the ignition lag of c.p. 80/20 FA-AN (see Table 1), but the same additive produced a marked decrease in the ignition delay of gasoline contaminated 80/20 FA-AN oxidized with WFNA having the same chemical analysis. In that connection, it should be pointed out that the shortest ignition delays obtained with additions of iron to the WFNA for the gasoline contaminated 80/20 FA-AN mixture were never less than the delays obtained with the pure fuel mixture.

Fig. 8 presents the ignition lags of furfuryl alcohol and of the 80/20 FA-AN fuel mixture oxidized with various mixtures of HNO_3 (NO_2 free) and water as a function of weight per cent of HNO_3 in the oxidizer mixture (7). It is noteworthy that the ignition delays obtained from furfuryl alcohol and from 80/20 FA-AN oxidized with 93.5 per cent HNO_3, 6.5 per cent H_2O, and no NO_2 were less than the delays obtained when these same fuels were oxidized with WFNA, the analysis of which was 93.5 per cent HNO_3, 6 per cent H_2O, and 0.5 per cent NO_2.

The reaction of aniline with various mixtures of HNO_3 (NO_2 free) and water produced a definite and reproducible spontaneous ignition only for acid concentrations in excess of 98.5 per cent; the ignition lag for aniline and an hydrous HNO_3 (NO_2 free) was determined to be 0.41 sec. However, the ignition lag of aniline oxidized with WFNA containing 93.5 per cent HNO_3, 0.5 per cent NO_2, and 6.0 per cent H_2O is about 0.25 sec (9).

On the basis of the aforementioned results it would appear that NO_2 in solution with the nitric acid improves the ignitability of the aniline-nitric-acid bipropellant system but is not beneficial in the case of the furfuryl alcohol-nitric acid and 80/20 FA-AN nitric-acid bipropellant systems.

The reaction of 6 millimeters of α-pinene with 12 milliliters of anhydrous HNO_3 (NO_2 free) produced sporatic spontaneous ignition with the ignition lags ranging from 0.5 sec to several sec; when the α-pinene was reacted with nitric acid-water mixtures of less than 98.5 per cent acid concentration, ignition did not occur although a violent frothing reaction was observed.

It has been pointed out that the purity of the propellants tested may influence the ignition lags. Therefore, it was necessary to control the chemical composition of the propellants tested. The procedure for controlling the chemical composition of the nitric acid is described in the following section.

Method of Preparation and Properties of Nitric Acid

The water-nitric acid mixtures used in the investigation described above were prepared in the following manner:

Potassium nitrate (c.p.) was reacted with sulphuric acid to form nitric acid in accordance with

$$H_2SO_4 + KNO_3 \rightleftarrows HNO_3 + KHSO_4 \ldots\ldots [1]$$

The nitric acid was removed from the products of the reaction by distillation under a pressure of approximately 35-mm mercury and a saturation vapor temperature range of 25 to 30 C. By maintaining a rela-

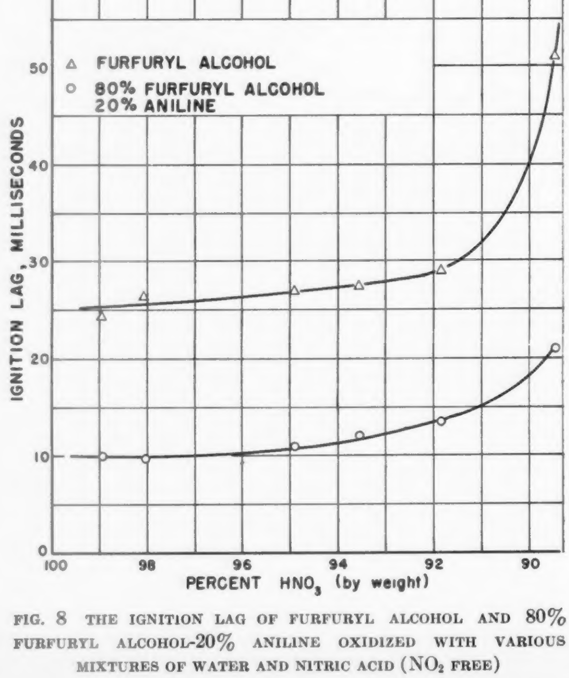

FIG. 8 THE IGNITION LAG OF FURFURYL ALCOHOL AND 80% FURFURYL ALCOHOL-20% ANILINE OXIDIZED WITH VARIOUS MIXTURES OF WATER AND NITRIC ACID (NO_2 FREE)

tively low distillation temperature, thermal decomposition of the nitric acid and hence formation of NO_2 were prevented. The distillate collected was purified by a series of fractional crystallizations (10). The separation produced an acid of 99 per cent purity and a eutectic mixture of the monohydrate ($HNO_3 \cdot H_2O$) and the pure HNO_3; the concentration of the eutectic mixture is about 90 per cent HNO_3 by weight. By blending the eutectic mixture and the 99 per cent acid in selected weight ratios, water-nitric acid mixtures of the approximate desired concentrations were obtained. All mixture samples were then analyzed for total acidity; the solidification temperatures of all mixtures were determined and the values obtained were in fair agreement with the data published by Kuster and Kremann (11).

Fig. 9 presents the freezing point curve for HNO_3-water mixtures; the curve was plotted from data obtained from (11) and (12), and the author checked independently several points on that curve.

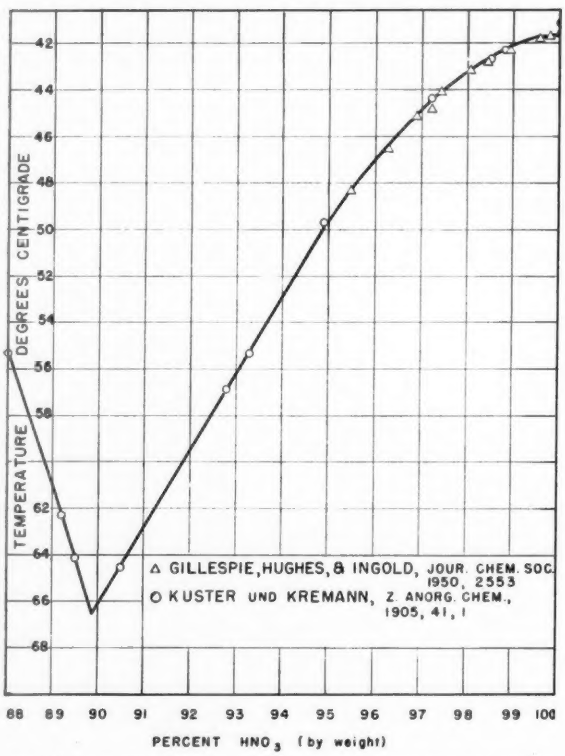

FIG. 9 FREEZING POINT CURVE FOR WATER-NITRIC ACID MIXTURE

Prior to being used, the aforementioned acid samples were stored in the dark in a refrigerator at a temperature of approximately −40 C. Under these conditions, the acid remained essentially stable for an extended period of time; in some cases for three months.

Conclusions

On the basis of the experimental data obtained from open-cup tests, it appears that the degree of liquid mixing achieved prior to combustion affects the ignition lag of the propellants concerned. The mixing of the propellants at low temperatures is impaired by the increased viscosity exhibited by the furfuryl alcohol and aniline, and it is believed that the poorer mixing is partially responsible for the excessive ignition delays which occur at low temperatures. However, even if the mixing were satisfactory, the ignition lag would be expected to increase due to the effects of the temperature coefficient of reaction rate, changes in surface tension, etc. The blending of the furfuryl alcohol and aniline in selected weight ratios will yield a fuel mixture which is less viscous at low temperatures than either constituent, and this effect is thought to be an important factor in the ability of the 80/20 FA-AN, WFNA bipropellant system to ignite smoothly and rapidly at temperatures somewhat less than the minimum temperature at which successful ignition may be expected from the furfuryl alcohol, WFNA bipropellant system. It may be possible, through the use of additives, to depress considerably the freezing point of a suitable blend of furfuryl alcohol and aniline without impairing the ignitability of the fuel mixture. The use of water-nitric-acid (NO_2 free) mixtures, in acid concentrations as low as 92 per cent, as oxidizers for furfuryl alcohol or the fuel mixture, 80 per cent FA, 20 per cent AN, appears to have little effect upon the ignition lags obtained with these propellants at room temperature, and the freezing temperature of the oxidizer mixture, 92 per cent HNO_3, 8 per cent H_2O, is about −75 F. Therefore the usefulness of the nitric-acid, furfuryl-alcohol, aniline propellant system might be extended to applications requiring successful hypergolic ignition for propellant temperatures as low as −60 F.

The ignition lag data obtained from the employment of the electronic ignition lag timer in open-cup tests have proved to be reproducible and accurate; in some cases, however, the purity and concentration of the propellants tested have had a discernible effect upon the ignition lags. Thus complete agreement between the ignition lag values reported by different investigators cannot be expected unless the chemical analyses of the propellants tested show that they had the same chemical composition. All too frequently ignition lag data are reported in the literature without listing the chemical analysis or concentration of the oxidizer. A considerable amount of valuable ignition lag data have been published, but the majority of the documents containing the data are classified. For that reason a comparison of the pertinent data of similar hypergolic bipropellant systems is difficult to make.

The correlation of ignition lag data obtained from open-cup tests and from actual motor tests has not yet been established (7); yet there is reason to believe that some sort of correlation exists. In any event, the open-cup test data are of value in screening propellants.

Acknowledgments

The information presented in this paper was obtained in conjunction with the work done at Purdue University under Phase 7, Project SQUID, a research program sponsored by the Office of Naval Research and the Office of Air Research, Contract N6ori-104, Task Order **1,** Phase 7. The author wishes to express his appreciation to Dr. M. J. Zucrow for his helpful guidance during the course of the investigations. Appreciation is also extended to Dr. W. L. Gilliland, of Gilliland Enterprises, for his co-operation as a consultant on pertinent problems of chemistry, and to personnel of the Purdue University Rocket Laboratory for their assistance.

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Letters to the Editor

This section of the Journal is open to letters not exceeding 600 words in length (or one and one-half columns) devoted to brief research reports or technical discussions of papers previously published. Such letters are published without editorial review, usually within two months of the date of receipt. The style and manner of submission of letters are the same as for regular contributions. (See inside back cover.)

On the Theory of One-Dimensional Flame Propagation

T. C. ADAMSON, JR.[1]

Guggenheim Jet Propulsion Center, California Institute of Technology, Pasadena, Calif.

Seemingly contradictory results have been obtained for the dependence of linear burning velocity on heat transfer to the flame holder (1, 2).[2] Although the analysis of Emmons and collaborators does not allow a solution involving zero heat transfer to the flame holder, the Hirschfelder, Curtiss, et al., development gives, explicitly, a solution for this case. Detailed analysis of this problem shows, however, that the results are different only because different mathematical descriptions of the cold boundary condition are involved. We shall demonstrate this result for flame propagation without diffusion,[3] using the notation of Hirschfelder, Curtiss and collaborators.

Assuming a unimolecular decomposition, $A \rightarrow bB$, and using the appropriate equations for conservation of mass and energy, it can easily be shown that

$$\lambda/M(d^2T/dx^2) = (w_A C_{P_A} + w_B C_{P_B} - \frac{1}{M}(d\lambda/dx))(dT/dx) + [(h_B - h_A)/M] M_A B^0 \eta_A \exp(-A^0/RT) \quad [1]$$

where λ is coefficient of thermal conduction; M, mass flow (ρv); T, temperature, η_A, moles of A per cc. of mixture; B^0, frequency factor; A^0, activation energy; C_{PA} and C_{PB}, heat capacity at constant pressure of A and B; h_A and h_B, enthalpy per gram of A and B; ρ density; and $w_A = M_A\eta_A/\rho$, $w_B = M_B\eta_B/\rho$, where M_A and M_B = molecular weight of A and B.

From [1] we can obtain two different results depending on the formulation of the cold boundary conditions. The assumption used by Emmons and collaborators,

$$(dT/dx)_c = (d^2T/dX^2)_c = 0 \quad [2]$$

where the subscript c stands for cold boundary, implies,

$$M = \infty \quad [3]$$

since $M_A B^0(\eta_A)_c \exp(-A^0/RT_c) > 0$ and $(h_B)_c - (h_A)_c \neq 0$ for $b \neq 1$, i.e., for the only type of unimolecular reaction which can lead to flame propagation. In fact, for all reasonable reactions $b > 1$ and $h_B < h_A$.

If it is assumed that

Received December 10, 1951.

[1] Guggenheim Fellow in Jet Propulsion.

[2] Refers to References at end of letter.

[3] The results are similar for flame propagation with diffusional transfer.

$$(dT/dX)_c = 0 \quad [4]$$

Then finite values are obtained for M if and only if

$$(d^2T/dX^2)_c = [((h_B - h_A)/\lambda)B^0\eta_A \exp(-A^0/RT)]_c \quad [5]$$

This is essentially the boundary condition adopted by Hirschfelder and collaborators for the case of no heat input to the holder. It should be noted that the limiting conditions involved in [3] and [5] can be obtained without confusing the problem by division of dw_B/dx by dT/dx for cases in which either or both of these derivatives vanish.

From the preceding discussion, it is apparent that the boundary conditions imposed by these two groups of authors differ, and hence the results also differ. Physically, we see that Emmons and his collaborators analyzed the case of premixed gases flowing through a channel, with a reaction taking place all along the duct and combustion occurring at the region of very great reaction rate. Hirschfelder and his group, however, analyzed the case of gases instantaneously mixed at the flame holder.

An alternate set of boundary conditions which may merit a detailed study is the following:

$$(d^2T/dX^2)_c = 0, \quad (dT/dx)_c \neq 0, \quad (dw_B/dx)_c \neq 0 \quad [6]$$

Equation [6] implies, for unimolecular decomposition and constant specific heats, the relation

$$(dT/dx)_c = [(1 - C_{P_B}/C_{P_A})T_cB^0/v] \exp(-A^0/RT_c) \quad [7]$$

if the initial gas mixture contains pure component A. These boundary conditions will also give finite flame speed.

The author wishes to give his thanks to Dr. H. S. Tsien and Dr. S. S. Penner for their helpful comments during the study of this problem.

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Jet Propulsion News

By C. F. WARNER, Purdue University, *Associate Editor*

with the assistance of W. G. BOHL

Rockets

BRITISH Armstrong-Siddeley Motors Ltd. has developed a 2000-lb thrust liquid propellant rocket motor named the "Snarler" for use as auxiliary power for jet aircraft. The unit, operating on a liquid oxygen and water-methanol mixture, weighs 215 lb and fits into a 3- × 6-ft space. The propellants are pumped from the supply tanks to the igniter and throttle valves, and the combustion chamber through 1-in. pipes by externally driven centrifugal pumps. The combustion chamber is regeneratively cooled by the water-methanol fuel mixture before it is injected into the combustion chamber. The Snarler has an endurance of three minutes at full thrust and has been test-flown in the Hawker P 1072 airplane.

✦ ✦ ✦

THE Swiss firm of Oerlikon, armament manufacturers, has produced a complete antiaircraft guided missile. The Oerlikon Machine Tool Works, Buehrle & Co., of Zurich, Switzerland, exhibited the missile and its control system at the Swiss National Air Display, Dubendorf Air Base, Zurich. The missile is about 16.5 ft long with a fineness ratio of about 12, and has four very thin wings indicating supersonic performance. A nitric-acid liquid fuel rocket motor with a duration of approximately 6 sec gives the missile a velocity of 2460 ft/sec at thrust cutoff. The 550-lb missile with a war head of approximately 45 lb may reach an altitude up to 66,000 ft. The guidance system is of the beam rider type with a range said to be about 12 miles. It is possible that a booster rocket may be used at launching.

✦ ✦ ✦

THE Army has awarded a contract to Firestone Tire and Rubber Co. to build a number of the Douglas-designed Corporal E tactical missiles. The missile is said to be about 40 ft long with a weight of approximately 12,000 lb, and has a probable range of 50 miles.

✦ ✦ ✦

ROCKET motor production by Ryan Aeronautical is being increased tenfold as a result of new orders received. Latest contract for missible motors is from Firestone Tire and Rubber Co. Previously Ryan had built rocket power plants for Douglas.

✦ ✦ ✦

SOME idea of the cost of modern jet units may be obtained from the fact that one take-off of a B-47 using 18 RATO units costs $5400, when auxiliary expenses are included.

Turbojet Engines

THE development by the Fairchild Engine Division of a turbojet engine of 1000-lb thrust was announced by

George F. Chaplin, vice-president of the Fairchild Engine and Airplane Corporation. The engine, designed and developed by Fairchild under contracts from the Navy and the Air Force, designated as the J-44, is approximately 6 ft long and 22 in. in diam. The total weight of the engine complete with accessories is 325 lb. It is also understood that the Fairchild Engine and Airplane Corporation's Stratos Division will manufacture the French Turbomeca's 160-hp gas turbine Oredon.

✦ ✦ ✦

CONTINENTAL Motors has entered the gas turbine field and will manufacture several small gas turbines under license from the French company Turbomeca. These units may be used to power small business and utility planes, target planes, guided missiles, helicopters, and liaison craft. These French turbines are reported to require less critical materials in their construction.

A partial listing of the Turbomeca's units is given in the following table.

Name	Output	Weight	Speed	Type
Aspin I	460-lb thrust	275 lb	Not listed	Ducted fan
Aspin II	730-lb thrust	300 lb	Not listed	Ducted fan
Artouste I	280 hp	185 lb	Not listed	....
Artouste II	400 hp	200 lb	Not listed	....
Palas	330-lb thrust	132 lb	34,500 rpm	....
Pimene	240-lb thrust	118 lb	35,000 rpm	....
Marbore II	815-lb thrust	Not listed	Not listed	....
Palouste	2.3 lb air per sec at 50 psia			turbocompressor

THE McDonnell Aircraft Corporation's "shortie" afterburner has become a well-paying part of the corporation's ramjet engineering program. Experimental contracts have been received to design and build these afterburners for the new model Allison and General Electric turbojet engines.

✦ ✦ ✦

RESULTS of an NACA research program at the Lewis Laboratory indicate that the temperature of gas turbine blades may be greatly reduced by the use of hollow-cooled blades. With a constant hot gas temperature of 1400 deg, a drop in blade temperature from 1400 deg for a solid blade to a temperature of 1100 for a hollow-cooled blade was obtained by the use of 1 per cent cooling air bled from the compressor. A further reduction of blade temperature to 775 deg was obtained by the use of 5 per cent cooling air. The new hollow blade has its cavity filled with small metal tubes brazed in place. This reduction in blade temperature means that it may be possible to increase the output of turbojet engines using high-temperature alloy cooled blades by increasing the gas temperature; or to obtain present-day outputs by the use of low-temperature alloy steel cooled blades, thus greatly reducing the required amount of critical materials. The NACA scientists also expressed the opinion that ceramal blades may soon replace present strategic materials in the manufacture of turbine blades. By the combined use of cooled blades and ceramals it may be possible to greatly increase the life of turbojet engines.

✦ ✦ ✦

A MAJOR subcontract to help build T-34 turboprop engines for new military aircraft has been let to Allis-Chalmers Manufacturing Company of Milwaukee, Wis., by Pratt & Whitney Aircraft. The T-34 Turbo-Wasp delivering both propeller power and jet thrust has ratings from 5000 to 6000 hp.

Aircraft

THE first Boeing B-47 Stratojet to be assigned to the U. S. Air Force Strategic Air Command has been delivered at MacDill Air Force Base, Tampa, Fla., by Boeing. Stratojets are also being built by Lockheed at Marietta, Ga., and by Douglas at Tulsa, Okla.

✦ ✦ ✦

McDONNELL's F-88 Voodoo has been ordered into production for the Air Force. The Voodoo has a span of 39 ft, 8 in., a length of 54 ft, a gross weight over 20,000 lb, and is powered by two Westinghouse J-34 WE 22 turbojet engines rated at 3600-lb thrust with afterburners. The sweptback wing Voodoo has a reported speed of over 700 mph and a range of over 1724 miles.

✦ ✦ ✦

SAPPHIRE turbojet engines built by the Wright Aeronautical Co. under license from Armstrong Siddeley Motors, Ltd., will furnish the power for the twin-jet English Electric Company's light bomber, Camberra, to be built by the Glenn L. Martin Company for USAF. The Sapphire, designated J-65-W-1, has a rated thrust of 7200 lb.

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PRESENTED in the photographs are three of the Delta-wing airplanes now undergoing extensive ex-

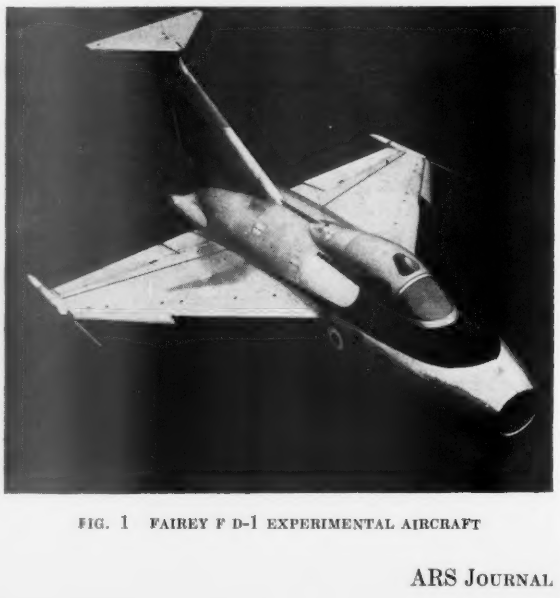

FIG. 1 FAIREY F D-1 EXPERIMENTAL AIRCRAFT

perimentation in England. No exact performance data are available at present; however, the approximate size and power of these aircraft are as follows: Fairey F D-1 smallest piloted Delta aircraft (Fig. 1) has a wing span of $19^1/_2$ ft and is powered by a Rolls-Royce Derwent turbojet engine rated at 3500-lb thrust. Boulton Paul P-111 (Fig. 2) has a 33-ft span and is powered by a Rolls-Royce Nene rated at 5000-lb thrust. The Avro 707A (Fig. 3) is $42^1/_2$ ft long, has a span of 34 ft, and is powered by a Rolls-Royce Derwent.

✦ ✦ ✦

THE new day-and-night jet fighter, the D.H. 110 (Fig. 4) is being flight tested by the de Havilland Company. The fighter, equipped with modern electronic navigation and combat aids, is powered by two Rolls-Royce Avon jet engines.

✦ ✦ ✦

AN UNCONVENTIONAL jet-propelled helicopter has been developed by the Rotor-Craft Corporation for the Office of Naval Research. The unit, known as the "pinwheel" by its makers, is a "knapsack" helicopter designed to drop armed troops behind enemy lines. Liquid fuel rocket motors are mounted at the tips of the two small rotor blades. A steel tube, to which the rotor is attached, forms the frame of the craft and carries the pilot's seat, fuel tanks, and cargo hook. The throttle-controlled rocket motors are said to be self-starting, emit no flame, and have a low fuel consumption, affording a reasonable combat range. The complete unit is said to weigh less than 100 lb.

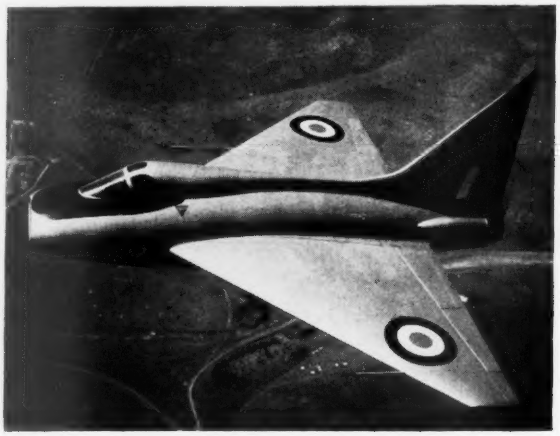

FIG. 2 BOULTON PAUL P-111 AIRPLANE WITH DETACHABLE WING TIPS FOR EXPERIMENTAL PURPOSES

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THE Navy has awarded a contract to McDonnell Aircraft Corporation to build a jet-propelled "cargo unloader" helicopter. The McDonnell design is said to use a single three-bladed jet engine driven rotor. McDonnell also hopes to be building its Little Henry Model 79 ramjet helicopters for crop-dusting work by spring.

✦ ✦ ✦

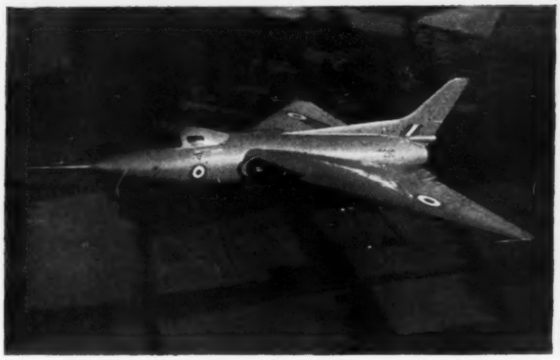

FIG. 3 AVRO 707A WITH EXTREME SWEEPBACK

HILLER Helicopter also plans to enter the commercial market with its two-place ramjet-powered Hiller Hornet.

✦ ✦ ✦

THE French are experimenting with a full-scale replica of a supersonic project at Orléans-Bricy and at Melun. The replica, in the form of a glider designated Arsenal 2-301, has a wing span of $29^1/_2$ ft with a thickness-to-chord ratio of 10 per cent and an over-all length of 41 ft.

New Facilities

GENERAL Motors plans to build two large double-duty plants near Willow Springs, Ill., and Arlington, Tex. Charles E. Wilson, president of GM, disclosed that the plants are designed to be able to shift from automobile production to jet engine production as the military and civilian production programs demand. The British-designed J-65 Sapphire turbojet engine will be assembled in the Willow Springs plant under American license from Curtiss-Wright.

✦ ✦ ✦

THE Swiss armament and machine tool manufacturer, Oerlikon Machine and Tool Works, Buehrle & Co. of Zurich, Switzerland, plans to establish an American subsidiary to be known as Oerlikon Tool and Arms Corporation of America. The company will specialize initially in the manufacture of aircraft armament and missiles. The new company, headed by Lt. Gen. (retired) K. B. Wolfe, former USAF deputy chief of

FIG. 4 NEW DE HAVILLAND JET FIGHTER

staff for materiel, has two other directors, Frederic Chapuisat, secretary and treasurer, and Leslie A. Skinner, retired colonel and developer of the Bazooka rocket gun, who is vice-president of engineering. Other members of the company include W. E. Tizzard, assistant to the president, and Clarence A. Davenport, demonstrator and test engineer. The American company will shortly announce plans to construct a major powder facility in the United States that will also serve as a final assembly point for various missile weapons. Initially the company will concentrate upon final assembly of component parts of weapons obtained from U. S. subcontractors for the military services.

The initial organization, operation, and research activity will be under the direction of the Swiss company. It is hoped that the American company will be financed and staffed completely by American personnel. After the company has become established it plans to produce a complete line of machine tools, electronic equipment, and business machines.

✦ ✦ ✦

A DIVISION for the research and development of helicopter jet engines has been established by Hiller Helicopter at its Willow Road plant, Palo Alto, Calif.

✦ ✦ ✦

JETLINER production has been halted by Avro because of Canada's extensive military commitments for Canuck jet fighters and Orenda turbojet engines. The company is closing its New York Office headed by R. Dixon Speas, who has resigned. Some development and testing work on the Jetliner is continuing.

✦ ✦ ✦

THE Joint Long Range Proving Ground established at Banana River, Fla., is nearing completion to its 1000-mile limit. The range, incorporating the former Naval Air Station at Banana River and the missile launching site at Cape Canaveral, is a test center only and has no facilities for research or development. The range spreads out over an area of 100,000 square miles to the southeast from Cape Canaveral and reaches its present limit of 1000 miles just beyond the southwestern tip of Puerto Rico. It is possible to extend the range to 5000 miles into the South Atlantic. At completion the range will have a launching site at Cape Canaveral and eight operating subdivisions or instrumentation stations between the launching site and the range limit. The Air Force has erected a concrete blockhouse to house instruments and observers, and will soon build a duplicate at the launching site so that two missiles may be fired in rapid sequence. The range officer obtains continuous information from constant course-plotting of the missile and can destroy it should it leave its correct course.

✦ ✦ ✦

THE new NACA supersonic tunnel costing 34 million dollars will be completed by next spring. This tunnel, the world's largest, will accommodate large long-range ramjets developing over 100,000-lb thrust at Mach numbers up to 3.5 at 40,000 ft. The present 8- by 6-ft supersonic tunnel has been used to test U.S.A.F. turbojets at an altitude of 65,000 ft at Mach 2, and ramjets have been tested at 80,000 ft. The drive for the new tunnel, a 250,000-hp unit, is being constructed in the Schenectady, N. Y., plant of the General Electric Company.

✦ ✦ ✦

IT HAS been reported that the U. S. Navy will erect a $30,000,000 guided missile plant at Bristol, Tenn., to be operated by the Sperry Gyroscope Company.

Personalities

BOEING Airplane Company has stationed a guided missile test group, headed by K. K. McDANIEL as field test director, at the U. S. Air Force Missile Test Center at Patrick Air Force Base, Fla.

✦ ✦ ✦

QUENTIN G. TURNER has been named manager of industrial engineering for Convair's guided missile division.

✦ ✦ ✦

CLAUDE N. MONSON, treasurer of the Garrett Corporation of Los Angeles, has been named manager of the AiResearch Manufacturing Company, a division of the Garrett Corporation. AiResearch has just been awarded a $36,000,000 contract by the Navy Bureau of Aeronautics for the production of small gas turbine engines used as turbojet and turboprop starters.

American Rocket Society News

By H. K. WILGUS, *Associate Editor*

Record Registration for Sixth Annual Convention of American Rocket Society

THE event of the year for scientists, engineers, and industrialists in the jet propulsion and rocket fields, the Sixth Annual Convention of the American Rocket Society, took place at the Chalfonte-Haddon Hall, Atlantic City N. J., on November 28–30, 1951. Held in conjunction with The American Society of Mechanical Engineers Annual Meeting, the total registration of 4500, of which 519 were ARS members and guests, surpassed former Atlantic City conventions.

The three-day American Rocket Society program was a full and stimulating one, consisting of the Annual Business Meeting, a Section Luncheon, Honors Night Dinner, and five technical sessions at which 16 papers were presented. The great interest in the achievements of the Society was shown by the constant activity around the publications booth and the large attendance at the technical sessions.

The ARS Annual Convention opened with its Annual Business Meeting on Wednesday morning, November 28. President H. R. J. Grosch opened the meeting, and reports were presented on the activities of the Board of Directors and the Society as a whole. The following officers were elected for the year 1952: *President*, C. W. Chillson; *Vice-President*, Lt. Comdr. F. C. Durant III, USN; *Directors* (for 3-year terms), G. Edward Pendray, Martin Summerfield, and Maurice J. Zucrow. An account of the Second International Congress on Astronautics held in London in September, 1951, was given by F. C. Durant III. The meeting concluded with a discussion of the progress and future plans of the Society. A detailed report on this meeting will be incorporated in the Annual Report to be distributed later.

First Section Luncheon Sets Precedent

At noon on Wednesday the American Rocket Society for the first time held a Section Luncheon for delegates and officers from all Sections and from localities where Sections are in the process of organization. Twenty-nine attended and the Sections were represented by the following members: *New York*, James Wheeler; *Indiana*, Clair M. Beighley; *West Texas-New Mexico*, Ralph F. Fearn; *Washington-Baltimore*, Harry J. Archer, Jr.; *Southern California*, C. C. Ross; *Niagara Frontier*, Kurt R. Stehling; and *Huntsville, Ala.*, A. L. Thackwell, Jr.

Dr. Grosch, who presided, referred to the outstanding developments in the ARS during the past year—the improvement in the JOURNAL, the increase in membership, and the strengthened financial position of the Society. He pointed out that the purpose of the Section Luncheon was to permit all Sections to report on their activities, and explain how the national organization can assist them further.

Harry J. Archer, Jr., president of the newly formed Washington-Baltimore Section, described the organizational meeting held recently in Washington, D. C. (A full account of this meeting will be found on page 46 of this issue.)

James W. Wheeler of the Sperry Gyroscope Company, president of the New York Section, announced that an active membership campaign is being initiated to increase the present number of 328. He reviewed the previous meetings held during the year, and made several suggestions for improvement of the liaison between the national organization and the Sections. He explained the advantages to be obtained by (1) exchanging of minutes of the National Board of Directors and Section Boards meetings; (2) establishing closer contact with the Sections so that papers presented can be distributed; and (3) listing in the JOURNAL a calendar of meetings and addresses of Secretaries for the benefit of members visiting other territories.

C. C. Ross, past-president of the Southern California Section, reported that the Section has held five meetings (two of which were closed), where the attendance exceeded 200. He stated that the concentration of military liaison officers in their territory simplified to some extent the question of clearance. Closed meetings were popular in southern California, he said, and

C. W. CHILLSON, CURTISS WRIGHT CORP., IS CONGRATULATED ON HIS ELECTION AS PRESIDENT OF THE AMERICAN ROCKET SOCIETY FOR 1952, BY THE HON. A. S. ALEXANDER, UNDER-SECRETARY OF THE ARMY (*center*) AND DR. H. R. J. GROSCH, RETIRING PRESIDENT OF THE AMERICAN ROCKET SOCIETY

DR. G. EDWARD PENDRAY, FOUNDER OF THE AMERICAN ROCKET SOCIETY, PRESENTING THE FIRST G. EDWARD PENDRAY AWARD TO MR. GEORGE P. SUTTON (*right*) FOR HIS BOOK "ROCKET PROPULSION ELEMENTS" DURING THE 6TH ANNUAL CONVENTION OF THE AMERICAN ROCKET SOCIETY AT ATLANTIC CITY, N. J.

probably would be in the East, also, if the problem of clearance was not insurmountable.

F. C. Durant III, in commenting on the closed meetings, said that although he realized the interest these meetings held for men in the rocket and jet propulsion fields, some consideration must be given to the fact they excluded student members who, he pointed out, are vital to the future of the Society.

Clair M. Beighley, past-president of the Indiana Section, explained the special problem that confronted the Section. Because the source of most of the membership is the student body at Purdue University, the turnover due to graduation and transfer makes the maintaining of a working group difficult. However, he said this problem is being solved by the establishment of a strong Board of Directors who will carry over the activities from year to year. He commended the untiring efforts of Maurice J. Zucrow who has been the guiding factor of the Indiana Section since its inception. Mr. Beighley also requested the national organization to consider a possible reduction in student dues.

Ralph F. Fearn, reporting for the West Texas-New Mexico Section, said that although activities had slackened in the past year, the Section undoubtedly could be stimulated to increase growth, and he agreed to work closely with the present officers to accomplish this.

Kurt R. Stehling represented the proposed Niagara Frontier Section which had held a first meeting recently with 400 members attending. Since there are a number of interested persons in that territory, he said, from such concerns as Cornell Aeronautical Laboratory, Buffalo Electro-Chemical, Carborundum, Bell Aircraft, etc., a well-established Section would probably soon affiliate with the American Rocket Society. He stated also that the Canadian Rocket Society appeared to welcome a tie-in with the American Rocket Society, depending upon how the affiliation could be arranged.

A. L. Thackwell, Jr., of Huntsville, Ala., expressed the belief that an active Section could be organized in that territory, and stated that he would make every effort to assist in such an organization.

C. W. Chillson, retiring vice-president of the American Rocket Society, remarked that it was Dr. Zucrow who first suggested the idea of a Section Luncheon, and that this suggestion should become the basis for a permanent meeting.

R. W. Porter, chairman of the Membership Committee, presented an encouraging picture of the growth of membership in the Society during the past year, and indicated on a map where Sections could be established in certain parts of the country not now under Section grouping. He considered it was essential for all ARS members to be in a specific Section, and a plan for redistributing the entire membership should be worked out.

Honors Night Dinner

One of the high points of the Convention was the Honors Night Dinner held in the Carolina Room on Thursday, November 29. Two hundred members and guests were introduced to C. W. Chillson, the incoming President of the American Rocket Society. Dr. Grosch, retiring President, in presenting Mr. Chillson, stated that the new President's claim to glory was not only because he would be the president for 1952 but also because of his untiring and successful work in building the technical programs for the Society for the past two years.

Dr. Grosch then welcomed the past-presidents of the Society: Alfred Africano, Roy Healy, James Wyld, Charles Villiers, John Shesta, and G. Edward Pendray. Also present to share in the welcome were S. Paul Johnston, Director of the Institute of the Aeronautical Sciences, and Ernest Hartford, Executive Assistant Secretary of The American Society of Mechanical Engineers, and Mrs. Hartford.

Awards Presented

In a brief ceremony, with Dr. Grosch making the presentations, Fellow Memberships were awarded to the following men for their outstanding contributions to the field of jet propulsion: Charles E. Bartley, Jet Propulsion Laboratory, California Institute of Technology; Benjamin F. Coffman, Jr., Department of the Navy; Edwin H. Hull, General Electric Company; Chandler C. Ross, Aerojet Engineering Corporation; and Colonel H. N. Toftoy, U.S.A.

Then followed the presentation of awards to men who have made significant contributions to rocket and jet propulsion research.

Because the two winners of the ARS Student Award, David Elliott and Leo Rosenthal of the California Institute of Technology, were unable to be present, the medals were handed to C. C. Ross, past-president of the Southern California Section, for presentation to the winners at a future Section meeting.

The G. Edward Pendray Award, presented for the first time for an outstanding contribution to the literature in the field, was given by Dr. Pendray to George P. Sutton, consultant, North American Aviation, Inc., for his book, "Rocket Propulsion Elements."

Dr. William H. Avery, Applied Physics Laboratory, The Johns Hopkins University, received the C. N. Hickman Award for his distinguished work in the field of solid propellant rockets. The medal

was presented by F. C. Durant III, in the absence of Dr. Hickman.

The Goddard Memorial Lecture Award was presented by Mrs. Robert H. Goddard to Comdr. R. C. Truax, U.S.N., for his pioneer experimental work on liquid propellant rockets.

Comdr. Truax Urges Promotion of Space Flight

In his acceptance speech, Comdr. Truax stated that the promotion of rocket development toward the goal of interplanetary travel was the highest objective the American Rocket Society could have. He said that satellite rockets capable of orbiting indefinitely around the earth are within the reach of present-day technology, and these rockets are the more useful of the immediately possible space-flight projects. He pointed out that such vehicles could be utilized in scientific investigations, in long-range radio communication, as a navigational aid, in certain military applications, and perhaps in many still unforeseen applications.

Truax suggested the appointment of a permanent space flight committee to promote the satellite project and solicit contributions from members and other interested activities for the required program of study, education, and promotion.

COMDR. R. C. TRUAX, USN, RECEIVING THE ROBERT H. GODDARD MEMORIAL AWARD FROM MRS. ROBERT H. GODDARD DURING THE 6TH ANNUAL CONVENTION OF THE AMERICAN ROCKET SOCIETY AT ATLANTIC CITY, N. J.

Haley Reports on IAF

Andrew G. Haley referred to Comdr. Truax's speech as a thorough expression of his own opinions regarding interplanetary travel, and stated that the subject was of vital interest to the American Rocket Society. Mr. Haley then reported on the establishment of the International Astronautical Federation at the Second Astronautical Congress held in London in September. Mr. Haley, who had been senior delegate from the American Rocket Society, was elected vice-president of the Federation for matters relating to the United States.

DR. WILLIAM H. AVERY (*right*), APPLIED PHYSICS LABORATORY, THE JOHNS HOPKINS UNIVERSITY, RECEIVES THE C. N. HICKMAN AWARD FROM LT. COMDR. F. C. DURANT, III, USN

Hon. A. S. Alexander Outlines Manpower Problem

The climax of the Honors Night Dinner was the address by the Hon. A. S. Alexander, Under-Secretary of the Army. Reviewing the aims and problems that the military forces and the rocket industry have in common, he urged that intensive research be done in all fields of rocketry, since the army has injected the rocket principle into practically every level of its weapons.

On the manpower problem, Mr. Alexander stressed the fact that the decline in the number of graduating engineers is a difficulty which the three services are trying to overcome. With more than half of the nation's scientific manpower being utilized in one way or another by the three services, something must be done, he stated, or great gaps will start to appear in our technological strength. He explained various methods by which the Army was attempting to train personnel and improve materials and operations.

He pointed out that missiles must be effective, transportable, and their operation easily understood if they are to be useful in modern combat. The frustrations have been many, but with every indication that guided missiles will soon be a major boost to our fire power, he considered that we have a right to be optimistic.

In closing, Mr. Alexander expressed the hope that the Society's current superlative job continues to be a major contribution toward a free world that is strong enough in every way to deserve peace, and that the not too distant future be devoted once again to the more satisfying objective of new attainments in speed and space.

EDITOR'S NOTE: For a review of the papers presented at the technical sessions, see pages 3–6 in this issue.

Washington-Baltimore Section Formed

THE Washington-Baltimore Section of the American Rocket Society was officially established at an organizational meeting held at the National Bureau of Standards on Wednesday evening, November 14, 1951. More than one hundred interested persons and members attended in spite of the rainy weather.

The meeting was conducted by the organizer and temporary chairman, Andrew G. Haley, of the National Board of Directors of the Society. Dr. H. R. J. Grosch, ARS President, gave a brief history of the Society and outlined the purposes and aims of a local chapter. He stressed the desirability and advantages of fostering social and professional contacts between the many people in the Washington-Baltimore area who are engaged in the wide range of scientific and engineering fields related to rocket propulsion and guided missiles. Dr. Grosch also expressed the hope that the program of this Section would include papers on various pertinent subjects which would not necessarily be completely ready for formal publication but which would serve to keep the members informed on the latest developments in the field.

Following Dr. Grosch's talk, Lt. Comdr. F. C. Durant III, USN, of the National Board of Directors, welcomed the members of the group. A set of By-Laws adapted to local conditions from the By-Laws of the New York Section was then presented by Marvin Hobbs. These By-Laws were accepted by the members without change. Nominations by a nominating committee and from the floor were presented to the members for a vote and the following officers were elected: *President:* Harry J. Archer, Jr., U. S. Naval Ordnance Laboratory, White Oak, Md.; *Vice-President* (Baltimore): William A. Webb, Aircraft Armament, Inc., Baltimore, Md.; *Vice-President* (Washington): Milton W. Rosen, Naval Research Laboratory, (Viking Program); *Secretary:* Miss Virginia R. Erwin, Consulting Radio Engineer, Washington, D. C.; *Treasurer:* Marvin Hobbs, Advisor to Chairman, Munitions Board; *Directors:* William L. Rogers, Aerojet Eng. Corp.; Charles F. Marsh, U. S. Government; Walter L. Webster, Jr., Baltimore; William A. Webb, Aircraft Armament, Inc., Baltimore; Irwin R. Barr, Washington, D. C.; J. R. Youngquist, Glenn L. Martin Co.; and Col. C. W. Eifler, U. S. Army.

After the election, a color movie on the Viking rocket was shown through the courtesy of Rear Adm. C. M. Bolster of the Office of Naval Research. Following the movie the new officers were introduced to the group and a general discussion of future programs was held under the new President of the Section.

The large attendance at this initial meeting, the enthusiasm and interest displayed during the discussions, and the interest of prominent military, Government and industrial personnel indicate that the Washington-Baltimore Section of the ARS has great potentialities of becoming a prominent technical and scientific association in this area.

Southern California Section Meeting Describes German Rocketry

A SUCCESSFUL dinner meeting, at which 350 attended, was held by the Southern California Section of the American Rocket Society on October 23, 1951, at the IAS Building, 7660 Beverly Blvd., Los Angeles, Calif. The chairman for the evening was Elmer P. Wheaton, project engineer for missiles, Douglas Aircraft Company, Inc.

Members and guests heard W. C. Noeggereth, formerly head of Rocket Propulsion Development, Air Research Institute, Munich, present a talk on "An Early Phase of Liquid Rocket Propulsion Development in Germany from 1935 to 1945." Dr. Noeggereth, now section head of the Underwater Ordnance Department, Naval Ordnance Test Station, explained the early basic investigations that had been conducted at the Institute on the new types of rocket propellants and their performance in the smaller scale rocket.

"Plan of Action for Use of Long-Range Ballistics Rocket A-4 (Later Called the V-2)" was the topic discussed by Walter Riedel, North American Aviation, Inc., formerly director of development and design at Peenemunde Proving Grounds in Germany. Dr. Riedel, in outlining the many difficulties that confronted the rocket industry in Germany, referred particularly to problems of production, handling, launching, and tactics.

The Southern California Section reports that the membership is now around 300, and that this type of dinner meeting is becoming increasingly popular.

At a meeting on December 12, 1951, at the Pasadena, Athletic Club, Pasadena, Calif., the following officers for 1952 of the Southern California Section were officially installed:

President, B. L. Dorman, Aerojet Engineering Corporation; *Vice-President,* W. J. Cecka, Jr., North American Aviation, Inc.; *Secretary-Treasurer,* G. D. Brewer, Hughes Aircraft Company; *Directors:* A. L. Antonio, Aerojet Engineering Corporation; R. B. Canright, Jet Propulsion Laboratory, Caltech; E. G. Crofut, Aerolab Development; S. K. Hoffman, North American Aviation, Inc.; C. McClosky, Office of Naval Research; R. C. Terbeck, Jet Propulsion Laboratory, Caltech; and D. A. Young, Aerojet Engineering Corporation.

The Section is scheduling its February meeting to be a confidential symposium on the physical aspects of solid propellants and the selections for airborne installations. The meeting, to be followed by a discussion period, is expected to parallel in interest the 1951 closed symposium on liquid propellants.

ARS Meets with Institute of the Aeronautical Sciences at Its 20th Annual Meeting

THE Institute of the Aeronautical Sciences will hold its 20th Annual Meeting on January 28 to February 1, 1952, at the Hotel Astor, New York, N. Y. A program of five technical sessions is planned in co-operation with seven other engineering groups: The Soaring Society of America, Institute of Radio Engineers, Radio Technical Commission for Aeronautics, Institute of Navigation, American Meteorological Society, American Helicopter Society, and the American Rocket Society. The five-day schedule will be concerned with such subjects as aeroelasticity, soaring, aerodynamics, electronics in aviation, meteorology, air transport, rocket and jet propulsion.

Members of the American Rocket Society will be particularly interested in the IAS-ARS rocket session and the IAS propulsion session to be held on Friday, Feb. 1. The tentative program is as follows:

9:00 a.m.–12:00 noon. Chairman, *C. W. Chillson,* Curtiss-Wright Corporation, President, American Rocket Society.

"Range Formulas for Rocket Powered Aircraft, "by *Ralph W. Allen,* head of Dynamics Structures Branch, Naval Air Missile Test Center, Point Mugu, Calif.

"Large Scale Production and Handling of

Liquid Hydrogen," by *H. L. Coplen*, senior engineer, Aerojet Engineering Corporation.

"The Role of Research in Rocket Development," by *Paul F. Winternitz*, director of research, Reaction Motors, Inc.

Luncheon. The guest of honor and principal speaker will be *Dr. Clark B. Millikan*, who is chairman of the Guided Missiles Panel, Research and Development Board, and director of the Guggenheim Aeronautical Laboratory, California Institute of Technology. Dr. Millikan will speak on guided missiles. Reservations for the luncheon should be made in advance through ARS or IAS. Price, $3.75.

2:00–5:00 p.m. Chairman, *S. T. Robinson*, of Sanderson & Porter, New York, N. Y.

(*Continued on page 53*)

Technical Literature Digest

By H. S. SEIFERT, California Institute of Technology, *Associate Editor*

CONTRIBUTORS: D. Altman, F. C. Gunther, D. I. Baker, S. S. Penner, R. B. Canright, H. S. Seifert, F. H. Wright

EDITOR'S NOTE:

The following collection of references is not intended to be comprehensive, but is rather a selection of the most significant and stimulating papers which have come to the attention of the contributors. The reader will understand that a considerable body of literature is unavailable because of security restrictions. We invite contributions to this department of references which have not come to our attention, as well as comment on how the department may better serve its function of providing leads to the jet propulsion applications of many diverse fields of knowledge.

Book Reviews

INTERPLANETARY FLIGHT, by A. C. Clarke, Harper & Brothers, New York, 1951, vii + 164 pp.

Reviewed by F. C. DURANT III

This work is highly recommended for the library of every professional engineer and scientists interested in or working in the field of rocket propulsion. This book should be there for two reasons. First, the initial reading is stimulating and enjoyable. Secondly, the book is a competent appraisal of major problems surrounding interplanetary travel—a venture which has been receiving increased thought and attention by many prominent persons connected with the guided missile field.

Mr. Clarke has examined the physical problems relating to the use of rocket power to master the earth's gravitational field and thus open the way to space travel. The reasons for use of satellite space stations as take-off points for interplanetary flights are developed with freshness and clarity. Atomic power for propulsion, orbital paths and velocities, mass ratios, space medicine, space suit requirements, meteor hazard, navigation and communication, and related subjects are treated in other chapters. In the final chapter, "Opening Frontiers," the author speculates soberly on the conquest of space and the impact which "astronautics" will have on mankind.

For the benefit of the casual or nontechnical reader, development of the mathematical relationships of the problems discussed are relegated to an appendix. The book is well illustrated with fifteen figures and fifteen photographic plates. This reviewer can find no criticism, unless it is the brevity of the work.

A few words about the author. Mr. Clarke has for more than ten years been an active member of the British Interplanetary Society, and is currently chairman of the B.I.S. Council. In September, 1951, he represented the Society as principal delegate to the Second International Astronautical Congress in London. As a scientist and as an author, he is particularly competent to chronicle the approach of interplanetary flight.

SPACE MEDICINE. Edited by John P. Marbarger, The University of Illinois Press, Urbana, 1951, 83 pp.

Reviewed by H. S. SEIFERT

This small volume is the outgrowth of a symposium on space medicine held at the Chicago campus of the Professional Colleges of the University of Illinois on March 3, 1950. It contains provocative discussions by biologists and physiologists on the many new medical problems posed by rocket flight. These include the effects of weightlessness, lack of orientation, solar radiations, oxygen supply, temperature extremes, and the like. Little more is done—or indeed at this time can be done—than to recognize and state the basic problems in a qualitative way. They are, however, intrinsically interesting and should stimulate further thinking and research.

A section by Wernher von Braun on the elementary principles of step-rockets and satellite rockets is lucidly written, and its contents will be familiar to most members of the ARS.

200 MILES UP, by J. Gordon Vaeth, The Ronald Press Company, New York, 1951, 207 + xiii. $4.50.

Reviewed by FRANK W. LEHAN

This book contains chapters on the physics of the upper atmosphere, instruments for astrophysical measurements, and high altitude test vehicles, including the "Skyhook" balloon, the author's specialty. It also treats qualitatively the basic principles of rockets, discusses procedures at White Sands Proving Ground, New Mexico, and goes into some detail concerning the V-2, Aerobee, and Viking rockets. It is written in nontechnical style.

The stated intent of Mr. Vaeth's book is to give the reader a picture of the United States' high-altitude research program which utilizes balloon and rocket flight. Particular emphasis is placed on the rocket vehicles, the various methods of instrumenting them and transmitting the data to the ground. The book appears to be successful in its aim, since it presents an entertaining and reasonably accurate account of our high-altitude research program. It describes in vivid detail many of the upper-atmosphere research problems, the instruments and methods used in investigating them, and the excitement of a rocket launching.

In accomplishing the above, the book also gives a glimpse into the U. S. rocket research program as a whole. The reader, however, should be aware that this glimpse is limited mainly to that small portion of the rocket program which has contributed to high-altitude research. Also, it gives principal emphasis to the Navy's aspects of the program, since Mr. Vaeth is associated with the Office of Naval Research.

Mr. Vaeth's perspective has led to a few statements which one with another viewpoint might rephrase. For example, it is stated (p. 117) that although now an upper-atmosphere research center, White Sands Proving Ground was organized primarily to serve as a temporary testing facility for small high-velocity rocket projectiles designed as weapons for ground and air forces. Actually, W.S.P.G. was organized by the Army as the first medium range U. S. guided-missile test activity, and the purpose of the majority of small projectiles fired there has been to check out the range instrumentation system or to provide basic data for the guided missile programs. It serves only incidentally as a high-altitude research center.

The book on the whole is clearly and authoritatively written and should provide informative reading for its intended audience.

Combustion, Flame and Explosions of Gases, by B. Lewis and G. von Elbe, Academic Press, Inc., New York, 1951, 795 + xix pp.

Reviewed by H. S. Tsien

If nothing else, the fact that there are 55 listings following B. Lewis and 49 listings following G. von Elbe in the author index of the book alone points to the great contribution by the authors in the field of combustion. Very few indeed would contest the recognition of the authors as the leading authorities on the subject chosen for the book. Since it is a rare event when a recognized authority finds time to write a book in his own field, the science and engineering public ought to be thankful when such an event actually happens, as in this case. There is no doubt about the up-to-dateness of this book, as many somewhat inaccessible unpublished reports are cited as references. In one instance at least, the subject matter of a paper published at approximately the same date as the book is included in the book (B. Karlovitz's investigation of turbulent flames)!

The book is divided into four parts: Part I gives the chemistry and kinetics of the reactions between fuel gases and oxygen (207 pp.). Part II gives the theory and the experimental data on flame propagation in laminar and turbulent streams, quenching, ignition, diffusion flames, and detonation (417 pp.). Part III treats the state of the burned gas, including its temperature and radiation (64 pp.). Part IV introduces the reader to the problems of engineering combustion processes (38 pp.). Various combustion data are assembled in three appendixes. This arrangement of material is thus logical and in the reviewer's opinion is superior to that of the older books by the present authors and W. Jost. In fact, aside from the conventional topics treated in Part III, the subject matter can be considered as arranged in order of increasing complexity, mostly complexity arising from the fluid mechanical aspects of the problem. One is perhaps right in saying that as one approaches the engineering combustion processes the fluid mechanical aspect of the problem becomes more and more important and, in the present stage of development of combustion theory, less and less is understood. This observation is attested by the brevity of the last part and by the fact that even this short section is less substantial than it appears. Half of it is engine cycle analysis and thus can be found in standard college textbooks on engineering thermodynamics.

It should be pointed out that the authors' approach to the complex problem of flame propagation is a semiempirical or phenomenological one. At the present stage of combustion theory, this is certainly the sound approach. The more "basic" approach, such as the theory of microstructure of combustion waves recently pursued so actively by J. O. Hirschfelder, is beset by serious difficulties in chemical kinetics, in boundary conditions, and in mathematical complexities. In the authors' opinion, "the important advances of combustion wave theory are reserved for the future" and "much of the former literature retains only historical interest."

In conclusion, the reviewer feels that the authors should be thanked for giving to this active field such a good and generally reliable reference book. For a research worker, it should be consulted constantly for information. For the student, it is a book in which to learn about the facts of combustion. In the face of such a major accomplishment, the reviewer feels that he should not try to point out the minor defects which should be obvious, after all, to the specialists on the subjects concerned.

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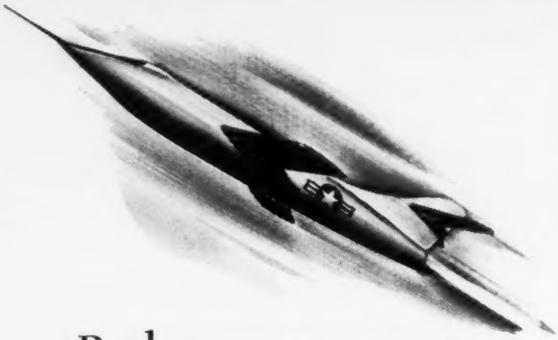

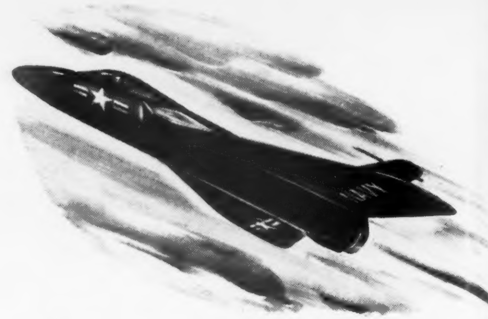

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